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OF A MODIFIED T-39 LIFT/CRUISE FAN V/STOL
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WIND TUNNEL AND GROUND STATIC TESTS
OF A .094 SCALE POWERED MODEL
OF A MODIFIED T-39 LIFT/CRUISE FAN V/STOL RESEARCH AIRPLANE

By D. Hunt, J. Clingan, V. Salemann and E. Omar

D180-20338-1

January 1977

Prepared under contract NAS2-9178 by

THE BOEING AEROSPACE COMPANY

BOEING MILITARY AIRPLANE DEVELOPMENT

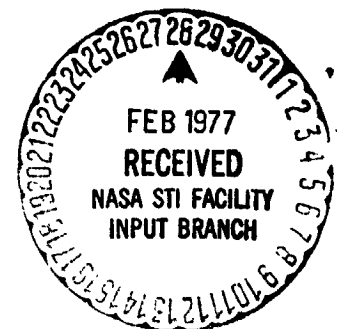
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16 Abstract This report presents the results of ground static tests and wind tunnel tests of a .094 scale model of a V/STOL research airplane. The configuration represents a modification to a T-39 airplane involving addition of a lift fan in the nose and replacement of the existing nacelles with tilting lift/cruise fans. The model was powered with three 14 cm diameter tip driven turbo-powered simulators. Forces and moments were measured by an internal strain gauge balance. Engine simulator thrust and mass flow were measured by calibrated pressure and temperature instrumentation mounted downstream of the fans. The tests were conducted to define the low speed handling qualities and general aerodynamic characteristics of the modified T-39. Test variables included: thrust level and thrust balance, forward speed, model pitch and sideslip angle at forward speeds, model pitch, roll, and ground height during static tests, lift/cruise fan tilt angle, flap and aileron deflection angle, and horizontal stabilizer angle. Tests were also conducted to determine the effects of removing the landing gear, the lift/cruise fans, and the tail surfaces.					
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FOREWORD

This report was prepared in fulfillment of National Aeronautics and Space Administration contract NAS2-9178. The contract was awarded March 30, 1976, to the Boeing Military Airplane Development Group of the Boeing Aerospace Company as part of a program to develop a Lift/Cruise Fan V/STOL Research Airplane.

The contract effort involved the design, fabrication, static ground testing, and wind tunnel testing of a .094 scale powered model of a modified T-39 Lift/Cruise Fan Airplane. Modifications to the T-39 included: expanding the forebody to house a lift fan, replacing the T-39 engines with tilting lift/cruise fans, raising the horizontal stabilizer to a T-tail position, and replacing the T-39 gear with A-4 gear. The model was powered by three 13.97 cm (5.5 inch) diameter turbo-powered simulators. Tests were conducted in the Boeing Static Test Facilities and in the Boeing 9-foot Low Speed Wind Tunnel. Forces and moments were measured by an internal strain-gauge balance. Engine simulator thrust and mass flow were measured by calibrated pressure and temperature instrumentation mounted downstream of the fan.

The tests were conducted to define the low speed handling qualities and general aerodynamic characteristics of the modified T-39. Test variables included: thrust level and thrust balance, forward speed, model pitch and sideslip angle at forward speeds, model pitch, roll, and ground height during static tests, lift/cruise fan tilt angle, flap and aileron deflection angle, and horizontal stabilizer angle. Tests were also conducted to determine the effects of removing the landing gear, the lift/cruise fans, and the tail surfaces.

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CONTENTS

	<u>PAGE</u>
1.0 SUMMARY	1
2.0 INTRODUCTION	4
3.0 NOMENCLATURE	6
4.0 TEST PROGRAM	10
4.1 Model Description	10
4.2 Test Facilities	14
4.3 Test Conditions and Procedures	16
4.4 Thrust Bookkeeping	22
4.5 Data Acquisition	23
5.0 TEST RESULTS	24
5.1 General Remarks	24
5.2 Static Hover Characteristics	26
5.3 Jet-Induced Aerodynamic Forces and Moments at Forward Speeds	31
5.4 Cross-Wind and Differential Thrust Effects	39
5.5 Configuration Component Effects	41
6.0 CONCLUSIONS	43
7.0 REFERENCES	45
FIGURES	47
APPENDICES	93
A - Wind Tunnel Test Data	94
B - Thrust Calibration	251
C - Data Reduction	290

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1.0 SUMMARY

A .094 scale powered model of a modified T-39 Lift/Cruise Fan V/STOL airplane was designed, fabricated and tested statically and in a low speed wind tunnel. The test program was conducted in fulfillment of NASA contract NAS2-9178, "Design and Fabrication of a Lift/Cruise Fan V/STOL Model for Wind Tunnel Tests", and was directed towards verifying the basic concept of a shaft-connected tilting lift/cruise fan V/STOL configuration and defining its general aerodynamic characteristics.

Modifications to the basic T-39 included extending the forebody to house the forward lift fan, replacing the T-39 engine pods with tilting lift/cruise fans, replacing the T-39 gear with A-4 gear, raising the horizontal stabilizer to a T-tail position, and providing a flap cutout door to provide clearance for the lift/cruise fan exhaust at high tilt angles. The model was powered by three tip-driven turbo-powered simulators. High pressure air to drive the simulators was supplied to the model through the sting support system and on through the internal "flow-thru" strain gauge balance.

The model was tested in four configurations. Namely:

- A VTOL configuration with flaps deployed, gear down, and engines tilted to 80° , 90° and 95° .
- A STOL configuration with flaps and gear down and engines tilted to 50° .

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- A CTOL configuration with flaps and gear down and engines not tilted.
- A loiter configuration with flaps and gear retracted and engines not tilted.

In addition, tests were conducted to assess the effects of removing the gear, nacelles, and tail surfaces and drooping the ailerons. Static tests of the hover configuration were conducted in the Boeing Static Test Facilities and test variables included model pitch, roll and ground height. All four configurations were tested at forward speeds in the Boeing 9' Low Speed Wind Tunnel at speeds ranging from 65 knots to 120 knots. Other test variables included model pitch, sideslip and thrust coefficient.

The static tests were directed towards defining the impact of ground effects upon hover lift and stability. The hover test results show an induced lift loss of 2% of gross thrust for heights far above the ground. With the model in a level attitude there was no additional lift loss due to ground effect at any height. The largest induced lift loss due to ground effect occurred at a combination of 10^0 of pitch and 10^0 of roll where a loss equal to 5% of gross thrust was measured. Instabilities in pitch and roll are present, however, the destabilizing moments are small relative to the control power available through differential thrust variation. The largest pitching moments and rolling moments found did not exceed 15% and 10% respectively of the control power available.

The emphasis during wind tunnel tests was on propulsion induced lift for STOL operation and the impact of the tilting lift/cruise fans upon longitudinal and directional stability. The useable propulsion induced lift of the STOL configuration was found to be 19% of the lift produced by the basic model with the lift/cruise fan nacelles removed. The induced lift from the T-39 model has been compared with that of a

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0.7 scale model of a lift/cruise fan configuration (Reference 1) which is based on thrust vectoring nozzles as opposed to tilting nacelles. The induced lift effects for the two configurations are quite comparable with the present configuration having somewhat greater induced lift for typical STO lift-off conditions.

The present combination of T-tail height and lift/cruise fan inlet length has a varying effect upon the longitudinal stability of the modified T-39 configuration. The configuration stability is very dependent on nacelle tilt angle, angle of attack, and thrust level. The longitudinal stability is adequate for the cruise configuration and degrades at some combinations of high nacelle tilt angle and high angle of attack. The control available using thrust transfer is more than adequate to handle the longitudinal moments. Horizontal tail control effectiveness is near the theoretical value for the cruise and CTOL configurations where the tail is required for control.

Directional stability is good in the VTOL mode at high angles of attack and degrades at lower pitch angles.

2.0 INTRODUCTION

NASA has undertaken the development of a Lift/Cruise Fan Research Airplane to provide a technical data base from which a Navy type "A" V/STOL aircraft and possible civil aircraft can be designed with confidence. The Navy requires a type "A" multi-mission V/STOL aircraft in the late 1980's which is capable of sea control operations from many small ships as well as ship-to-shore and shore-to-ship functions. The Lift/Cruise Fan concept exhibits an excellent potential for a type "A" aircraft because of its high speed, high altitude and range capabilities coupled with its ability to operate from small ships.

As a part of the NASA program, a contract was awarded to the Boeing Company to design, fabricate, and wind tunnel test a powered model of a modified T-39 Lift/Cruise Fan Airplane based on a tilt nacelle concept. The objective of the test program was to define the handling qualities of the T-39 configuration during hover in ground effects and at forward speeds out of ground effects. Static ground tests were conducted in the Boeing Static Test Facilities. Wind tunnel tests were conducted in the Boeing 9' Low Speed Wind Tunnel. Four configurations were tested. These include:

- A VTOL configuration with flaps deployed, gear down, and engines tilted to 80° , 90° , and 95° .
- An STOL configuration with flap and gear down and engines tilted to 50° .
- A CTOL configuration with flaps and gear down and engines not tilted.
- A loiter configuration with flaps and gear up and engines not tilted.

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In accordance with the terms of NASA contract NAS2-9178, this document is submitted as the final technical report. Presented herein is a description of the contract test program and of the analysis of the test results. The data acquired during the contract test program are included in Appendix "A". Details of the thrust calibration of the turbo-powered simulators are included in Appendix "B". Data reduction details are included in Appendix "C".

3.0 NOMENCLATURE

The ensuing list of definitions in most instances include two symbols. The first is the conventional symbol while the second represents a computer name appearing on the machine plotted data of Appendix "A". Units are presented in the SI system followed in parenthesis by the British Gravitational units.

- b, B - wing span, centimeters (inches) = 127.3 (50.11)
- \bar{c} , C, MAC - mean aerodynamic chord, centimeters (inches) = 24.02 (9.46)
- C_J , CJT - thrust coefficient = F_G/qS . CJA, CJB, and CJC are the thrust coefficients of each individual propulsion unit Fan A, Fan B, and Fan C (see Figure 2) " C_J = RAM" is defined in Section 5.3
- CFR - ram force coefficient = ram force/ qS . CFRA, CFRB, and CFRC are the ram force coefficients of each individual propulsion unit

The following group of stability axis coefficients includes all forces and moments on the model (i.e., aerodynamic, thrust, and ram).

- C_L - lift coefficient = L/qS
- C_D - drag coefficient = D/qS
- C_{PM} - pitching moment coefficient = $PM/qS\bar{c}$
- C_{SF} - side force coefficient = SF/qS
- C_{YM} - yawing moment coefficient = YM/qSb
- C_{RM} - rolling moment coefficient = RM/qSb

This next group of stability axis coefficients and ratios ending in "A" denotes aerodynamic forces and moments. That is, the direct thrust and ram forces and moments have been removed (see Figure C.1).

C_{L_A} , CLA	- aerodynamic lift coefficients = L_A/qS
C_{D_A} , CDA	- aerodynamic drag coefficient = D_A/qS
C_{PM_A} , CPMA	- aerodynamic pitching moment coefficient = $PM_A/qS\bar{c}$
C_{SF_A} , CSFA	- aerodynamic side force coefficient = SF_A/qS
C_{YM_A} , CYMA	- aerodynamic yawing moment coefficient = YM_A/qSb
C_{RM_A} , CRMA	- aerodynamic rolling moment coefficient = RM_A/qSb
L_A/F_G , LA/FGT	- aerodynamic lift ratio
D_A/F_G , DA/FGT	- aerodynamic drag ratio
$PM_A/F_G\bar{c}$, PMA/FGT*C	- aerodynamic pitching moment ratio
SF_A/F_G , SFA/FGT	- aerodynamic side force ratio
YM_A/F_Gb , YMA/FGT*B	- aerodynamic yawing moment ratio
RM_A/F_Gb , RMA/FGT*B	- aerodynamic rolling moment ratio
D	- drag, newtons (pounds)
D_{HI}	- highlight diameter, centimeters (inches) = 15.66 (6.164)
F_G , FGT	- total gross thrust of all three propulsion units = $F_{GA} + F_{GB} + F_{GC}$, F_{GA} , F_{GB} and F_{GC} represent the gross thrust of each individual propulsion unit, newtons (pounds)
FPR	- fan pressure ratio = P_{T_3}/P_0
h , HGT	- height of model reference point above ground plane (see Figure 2), centimeters (inches)

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h/\bar{c} , HGT/C	- ratio of model ground height to mean aerodynamic chord
L	- lift, newtons (pounds)
P_{T3}	- total pressure downstream of fan, newtons/meter ² (psf)
P_o	- wind tunnel test section static pressure, newtons/meter ² (psf)
PM	- pitching moments, newton meters (ft-lbs)
q, Q	- wind tunnel test section dynamic pressure, newtons/meters ² (psf)
RM	- rolling moments, newton meters (ft-lbs)
S	- wing reference area, centimeters ² (in ²), = 2808 (435.17)
α_i	- incidence of horizontal tail - degrees
SF	- side force, newtons (pounds)
V/V_j	- ratio of freestream velocity to fan efflux velocity
YM	- yawing moments, newton meters (ft-lbs)
α , ALPHA	- angle of attack of body axis, degrees
β , BETA	- angle of sideslip of body axis, degrees (positive nose left)
λ , LAMBDA	- lift/cruise fan tilt angle relative to body axis degrees
ϵ_α	- downwash gradient at horizontal tail, $dc/d\alpha$ (ϵ positive for downflow at tail)
ϕ	- angle of roll relative to ground plane (positive right wing down), degrees

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- Notes: (1) The moment reference center is at the station of $.3\bar{c}$, F.S.
60.95 cm (23.994) and at water line 22.68 cm (8.93 in).
- (2) The sign convention for forces and angles is given in
Figure C.5 of Appendix C.

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4.0 TEST PROGRAM

4.1 Model Description

The .094 scale model represented a V/STOL technology demonstrator airplane based on a modified T-39 (Model 1041-135-2R of Reference 2). The modifications include:

- Expanding the forebody to house a lift fan
- Replacing the T-39 engine pods with tilting lift/cruise fan pods
- Raising the stabilizer to a T-tail position
- Replacing the gear with A-4 gear
- Adding a flap door to provide clearance for the lift/cruise fan efflux when the nacelles are tilted to the hover position.

The model was sting-mounted, which necessitated some distortion of the aft fuselage to allow sting clearance (see Figure 1), and was supported on a six-component strain-gauged, "flow-thru" balance (illustrated in Figure 3 and discussed in Reference 3).

Figure 2 shows the model in its various configurations and gives the principal dimensions. The top view depicts the CTOL configuration, with the nacelles at $\lambda = 0^\circ$ and the front fan doors open; the side view depicts a VTOL configuration, showing the nacelles at $\lambda = 90^\circ$. The flaps and flap doors are shown extended in the VTOL configuration.

Three Technology Development Inc. 14 cm (5.5 inch) diameter tip driven fans were used to simulate the propulsion system. One of these units is shown, disassembled, in Figure 4 and a description is found in Reference 4. The units were furnished by NASA for this test program.

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Each fan was rated at 35000 RPM with a design fan pressure ratio of 1.25. At this rating each had a fan airflow of 2.5 kg/sec (5.55 lbs/sec) and used .5 kg/sec (1.1 lbs/sec) of drive air. For the purpose of this test, they were operated up to a maximum of 30000 RPM. The drive air was supplied to the model at 41.4 bars (600 psig) and heated to 71°C (160°F) to prevent icing at the turbine exit. Remotely controlled valves and critical-flow venturi meters located within the model were used to control and monitor the airflow to each individual fan (Figure 5).

The high pressure air supplied to the L/C nacelles was divided into two portions, one part for driving the fan and the other for simulating the primary exhaust efflux. Since both portions were always choked the simulated primary weight flow varied in direct proportion to the fan supply weight flow. The primary air was passed through a choke plate and screens to obtain a uniform total pressure distribution and representative pressure ratio before exhausting from the primary nozzles. Figure 6 shows details of the lift/cruise nacelle assembly. Each nacelle can be pre-set to various tilt positions, representing various flight configurations. A serrated interconnection allowed angular settings in increments of 5°. The limits of angular rotation were determined by instrumentation cables and tubes which were routed out of the nacelle around the periphery of this connection. This allowed rotation from $\lambda = 0^\circ$ (aligned with the model axis) to $\lambda = 105^\circ$, with the pivot point located at fuselage station 82.93 cm (32.65 inches) and water line 30.53 cm (12.02 inches).

The external lines of the nacelle were generated to represent those of the technology demonstrator having Allison PD 370-16 engines installed. The model scale was selected such that the fan exit area was scaled exactly. A more complete description of the design process is included in Appendix "B" which describes how the inlet contours were chosen and gives details of the nacelle ordinates. Appendix "B" also describes the instrumentation in the nacelle. The external cross-section of the core engine cowl was elliptical as indicated by the shading in Figure 6.

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This was done on the assumption that the engine accessories could be distributed between the top and bottom of the core engine.

Fairings were used to cover the Lift/Cruise Fan nacelle pivots, instrumentation connections and air supply lines. These fairings are not completely representative of the full scale airplane. Part of each fairing was attached to, and rotated with, the nacelle and part was attached to the fuselage. These fairings are shown detailed in Figure 7. In order to determine the aerodynamic effect of the fairing attached to the nacelle, it was removed for one test and the leading edges of the nacelle fittings were faired with modelling clay as shown in Figure 7.

Yaw vanes were located at the fan nozzle exit for thrust deflection and are shown in their undeflected position in Figure 6. The vane chord was 35.56 cm (14 in) full scale and its section was that of a NACA 0012 airfoil. The pivot point was at 10% of vane chord with the vane leading edge located at the fan exhaust plane. Deflection angles of 0° , $\pm 10^{\circ}$, and $\pm 20^{\circ}$ were provided.

Details of the nose fan installation are given in Figure 8. The nose fan was located as low in the fuselage as possible in order to allow length for development of the inlet contours. The fuselage forebody was hand-worked to provide the desired inlet shape. Sections of the inlet are presented in Appendix "B".

The nose fan inlet doors, which when closed form the upper surface of the airplane nose, are shown diagrammatically on Figure 8 in the open and closed positions. When the doors were in the open position, they were stowed at the sides of the fuselage as shown. Parts were provided to test both open and closed configurations.

The nose fan exit doors which, when open, can be deflected to provide thrust vectoring for yaw control are shown in the open position in Figure 8. The outboard doors opened towards the airplane plane of symmetry and the inboard doors opened from that plane outwards to form

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the projections shown. A disc was provided to fair and seal the exit when a "door-closed" configuration was tested.

The basic T-39 model airframe was defined in accordance with the Saberliner Specifications Document (Reference 5) with additional consideration of measurements taken from a full scale T-39. Wing and tail planforms and section definitions are presented in Figures 9 and 10. The model wing was positioned .38 cm (.15 inches) lower relative to the fuselage than full scale airplane definitions indicated. This had the effect of increasing the body depth in the vicinity of the wing and was done to provide more space within the model for structure and instrumentation. The wing dihedral was 3.15° . The wing was twisted linearly from zero incidence at the root to -2.9° at the construction tip. The tip of the vertical fin was modified to support the horizontal stabilizer. A NACA-0010 airfoil section was used for the vertical fin. The airfoil section of the all new horizontal stabilizer varied linearly from a NACA-0010 airfoil at the tip to a NACA-0015 airfoil at the root. The stabilizer had an anhedral of 9° .

The basic high lift system (i.e., slats and flaps) were positioned in accordance with measurements of a full scale airplane in their fully deployed position. The slats and flaps were unchanged throughout the test with the exception of tests of the loiter configuration at which time the slats and flaps were nested. The flap doors, designed to provide clearance for the lift/cruise fan efflux, were deflected for tests of the hover configuration when λ was 90° or greater and retracted for all other tests.

No transition grit was used on the model for any of the tests reported herein.

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4.2 Test Facilities

All tests were conducted in Boeing facilities located in Seattle, Washington. The facilities involved are described briefly below.

4.2.1 Flight Simulation Chamber (FSC)

Preliminary calibrations of the model fans were conducted in the Boeing Wind Tunnel Flight Simulation Chamber. This facility was developed over a period of years specifically for the purpose of calibrating wind tunnel model propulsion simulators. The facility includes thrust and mass flow measuring systems and allows for variation of back pressure by means of evacuating the chamber into which the thrust simulator discharges. A schematic of the chamber is shown in Figure 11. A detailed description of the facility is presented in Reference 6.

4.2.2 Boeing Static Checkout Area

All of the static hover tests were conducted in this facility. The primary use of this area is for checkout of models being prepared for tests in the Boeing Transonic Wind Tunnel. This facility is essentially a 8.5m x 12.2m by 4.0m high room (28 ft x 40 ft x 13 ft) as shown in Figure 12, equipped with a model support system providing for pitch and vertical translation of sting mounted models. The model is controlled from an adjacent control room with visibility provided by both a window and closed circuit television. Instrumentation outputs are carried to the Boeing Supersonic Wind Tunnel Data System by means of a permanent hardwired interconnect. Both on-line and final data reduction are accomplished by the wind tunnel central computing system. The model was installed in approximately the center of the room with the model axis aligned with the longer (12.2m) room dimension. The room floor served as the ground plane.

4.2.3 Boeing 9'-B Low Speed Wind Tunnel

The wind tunnel testing was conducted in the 9' -B Low Speed Wind

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Tunnel located at the Propulsion Laboratories' North Boeing Field Test Complex. The 9'-B LSWT is an open circuit wind tunnel which draws air directly from the atmosphere through a test section 2.59 meters (102 inches) high by 2.67 meters (105 inches) wide. A schematic of the tunnel is shown on Figure 13 and a view of the model installed in the tunnel is shown in Figure 14. The tunnel is powered by an Allison model 501-D13 gas turbine engine using a variable pitch propellor. Flow straighteners and screens are located in the tunnel inlet to minimize the effect that atmospheric wind conditions have on the tunnel flow profile. Tunnel velocities can be varied from 0 to approximately 180 knots. The tunnel is equipped with a sting support system mounted in the aft portion of the constant area section of tunnel. The motion of the support system is in the horizontal plane such that the model wings are in the vertical plane as shown in Figure 14.

The 9'-B LSWT Data System was used for recording the data on magnetic disc. The Acquisition System is a software controlled, Boeing designed Standard Digital Data System (SDDS). A PDP-8 computer is used for system control and on line "quick-look" data calculation with CRT output. A PDP-8/I is used for on/off-line final data calculation.

4.3 Test Conditions and Procedures

4.3.1 Test Conditions

The model was tested in four basic configurations. These are defined as follows:

- A VTOL configuration with flaps deployed, gear down, and engines tilted to 80° , 90° and 95° .
- A STOL configuration with flaps and gear down and engines tilted to 50° .
- A CTOL configuration with flaps and gear down and engines not tilted.
- A loiter configuration with flaps and gear retracted and engines not tilted.

The conditions under which each was tested are summarized in Figure 15. Included are lift/cruise fan tilt angle, forward speed, thrust, pitch and sideslip angle, stabilator angle as well as component effects. In addition, the VTOL configuration was tested with differential thrust, side-to-side and fore and aft, representative of thrust control variations. Only the VTOL configuration was tested statically. Test conditions for the static test included: engine tilt angles of 80° , 90° , and 95° ; pitch angles of -5° , 0° , 5° , 10° , and 15° ; roll angles of -5° and -10° ; ground height variation from .12 meters (4.7 inches) to 1.83 meters (72 inches).

4.3.2 General Procedures

The model fans were supplied with high pressure air from a plenum inside the model. This plenum is essentially part of the internal balance and the pressure level in the plenum has some effect on the balance outputs. The balance plenum pressure was always set and maintained at

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a level of 41.4 bars (600 psig) for all tests, fan calibrations, and balance calibrations. This procedure is standard with this type of balances and insures that the measured force levels are not influenced by balance pressure tares. Since the individual fan control valves are downstream of this plenum there is still complete control of fan thrust without varying the balance plenum pressure.

The high pressure air used to drive the fans was pre-heated to eliminate icing problems. During the initial calibration it was determined by observation of the fans that the minimum supply air temperatures at which visible ice would not form was about 71°C (160°F) as measured in the balance plenum. This temperature was used for all further testing. In the colder environment of the open circuit wind tunnel, it was infrequently observed that under some conditions ice could be seen on the fan shroud between stator blades. This condition would only last for a few minutes when the ice would fall off.

Since warmed air was being carried across the internal balance it was necessary to establish procedures to minimize balance zero drift. During the static test, it was found that by pre-warming the balance to about 54°C (130°F) prior to recording zeros, the problem of balance output drift was minimized. However, during the wind tunnel test it was not possible to maintain the balance at this temperature during a run because the heat input from the warmed air was not adequate in the presence of the large heat transfer from the model to the tunnel stream. This resulted in greater thermal gradients and consequently greater balance output drifts than those occurring during the static tests.

A serious problem in measuring fan exit total and static pressures occurred during the static test. Erratic readings were observed and were eventually traced to an accumulation of fan lubricating oil in the pressure measuring lines. This problem is apparently related to the large quantity of oil-mist which must be supplied to the fan bearings. The oil-mist is blown into the bearing cavity and subsequently flows out

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into the fan air stream ahead of the pressure instrumentation. Because of this problem, RPM rather than pressures was used to reduce the static data. For the wind tunnel data it was not considered adequate to rely on RPM in the presence of the substantial levels of fan flow distortion and back pressure variations expected during forward speed operation at large nacelle tilt angles. The pressure measuring problem was eliminated for the wind tunnel tests by replumbing the model to utilize external rather than internal pressure scanning valves. This allowed the use of an existing system which is designed to blow dry nitrogen from the pressure scanner back through the pressure lines to remove any foreign matter. This is accomplished by an automated system during the off-shift and on this test every pressure measuring line was blown out after each day's testing. This procedure was sufficient to solve the problem since some time was required to accumulate sufficient oil to cause a problem.

4.3.3 In-Place Fan Calibrations

During both the static and wind tunnel tests, the fans were calibrated in place.

In the static test facility this calibration consisted of positioning the complete model at about 1.8 meters (72 inches) above the floor and at an angle of attack of 15° to minimize the ground effects within the constraints of the support system. Each fan was then run separately (lift cruise fans tilted to their 90° position) to determine the relationships between RPM and thrust as measured by the internal force balance. (Thrust was also computed from fan pressure instrumentation but due to instrumentation difficulties this data was not utilized for the static test data reduction.) The fans were run at thrust levels up to about 400 newtons (90 lbs).

In the wind tunnel, similar calibrations were run with the exception that the fan efflux was directed into a scavenging pipe which directed it outside the test section. This prevented re-circulation of the fan flow around the model and also prevented the inducement of a flow

velocity through the wind tunnel by the ejector action of the fans. The pressure instrumentation difficulties of the static test had been solved and the calibrations were used to relate the forces measured by the internal balance to those computed from the fan exit pressure instrumentation.

The above calibrations established the basis for determining thrust during actual tests. In addition, a variation of thrust coefficient with back pressure was included in the data reduction. This relationship had been established during the initial calibration of each fan in the Flight Simulation Chamber.

The calibrations are described in greater detail in Appendix "B".

4.3.4 Static Test Procedure

For the static testing the model was initially positioned with the landing gear clear of the ground plane by about .25 cm (.1 inch). Each fan was set at a specified RPM value selected to give the desired thrust level. The lift/cruise fans were generally set to give about 350 newtons (79 lb) thrust each. This amount of thrust corresponds to a fan pressure ratio of 1.14 which matches the full scale airplane value. The nose fan was set to give an approximate pitching moment trim about the moment reference center. This value was computed from geometric considerations and varies with nacelle tilt angle as shown in Figure 16. Certain runs were also made with the thrust intentionally unbalanced, either side to side or fore and aft.

The model was then traversed away from the floor with periodic stops for pressure stabilization and data recording. Constant angle of attack was held throughout a height traverse. Due to ground effects, the thrust levels (especially of the nose fan) tended to change somewhat with height above the floor. No attempt was made to maintain the original thrust levels. The control parameter was tip turbine supply pressure which was maintained constant during height traverses.

4.3.5 Wind Tunnel Test Procedures

In general the desired variations in thrust coefficient (C_J) were obtained by changing the tunnel speed at a constant fan thrust setting. The predominant thrust setting used corresponded to a fan pressure ratio of about 1.14. The tunnel speeds run were in the range 33 to 62 m/sec (65-120 kts) giving a C_J range of about 1.8 to 5.5 and corresponding to a Reynolds number range from .6 million to 1.2 million based on the model reference chord. However, partway through the test a model structural problem was discovered which required limiting the tunnel speed to about 36 m/sec (90 kts) compared to the previously selected maximum of 62 m/sec (120 kts). Thereafter the lowest C_J value ($C_J = 1.8$) was run at a fan pressure ratio of about 1.10 while for all higher C_J values the FPR = 1.14 setting was maintained.

The upper C_J limit of 5.5 was selected based on considerations of tunnel flow breakdown. While time did not permit a detailed study of that limitation, observations were made of tufts on the tunnel walls at several fixed angles of attack as C_J was gradually increased. These runs were made with the nacelle tilt angle set at 90° . It was observed that the flow on the tunnel wall beneath and behind the model became increasingly rough as C_J increased. At $V = 33$ m/sec (65 kts, $C_J = 5.5$) the region of rough flow had moved forward to a position under the model tail but there were no significant areas of reversed flow. However, when the speed was lowered to about 31 m/sec there were large areas under the model where the flow was actually reversed. This was interpreted as a "flow breakdown" situation representing a region where valid free air testing was not possible. In no case was the tunnel flow observed to "climb the sidewalls" as has been observed by others for flow breakdown situations. It is worth noting that the test limit determined by this method is in good agreement with a detailed flow breakdown study presented in Reference 7.

The thrust balance among the three fans was generally set to provide approximate moment trim about the moment reference center for the

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static case. No attempt was made to readjust the initial static supply pressure values to allow for thrust changes or aerodynamic moments resulting from forward speed.

In general, angle of attack series were taken with angle increasing from -8° to $+30^{\circ}$. It was noted that some hysteresis existed and that for certain configurations this effect was appreciable. This phenomenon was not explored in any detail and in general the data presented are all on the same side of the hysteresis loop.

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4.4 Thrust Bookkeeping

The thrust subtracted from the measured forces was calculated differently in the resolution of wind tunnel data compared to static test data. In the static test, nozzle total pressure data were not reliable, and thrust was obtained from a calibration against fan corrected speed. In the wind tunnel, pressure data were reliable and thrust was calculated from nozzle total and static pressure measurements. As a result, full scale engine thrust calculations must be performed differently when they are to be combined with model wind tunnel aerodynamic data as compared to static ground effects data.

When applied to wind tunnel data, full scale thrust must reflect the effect of nozzle back pressure on engine performance. This requires knowledge of back pressure vs. nacelle attitude and airspeed, and the response of the engine to back pressure changes.

During static testing, corrected speed did not vary with height, therefore calculated thrust did not vary either. However, there was some evidence, particularly on the nose fan, that nozzle back pressure varied in ground effect. Any effect on actual fan thrust would therefore show up as an aerodynamic ground effect on the airframe. Therefore, where static model data is used for full scale predictions, it should be combined with full scale thrust out of ground effect. The result is valid if the model and full scale fans react similarly to increasing nozzle back pressure.

4.5 Data Acquisition

The data acquired during the wind tunnel test program included:

- Model force and moment measurements (axial force, normal force, side force, and moments about the pitch, yaw and roll axis) from an internal, flow-thru, strain gauge balance.
- Balance temperature as well as the temperature and pressure within the balance plenum.
- Jet efflux total pressure, static pressure and total temperature.
- Pressure and temperature of the individual air supplies to power the three turbo-powered simulators.
- Simulator inlet total pressures at the forward (or lower in the case of the lift/cruise fans when not tilted) azimuth as well as inlet static pressure.
- Simulator RPM and bearing temperature. The bearing temperature was monitored but not recorded.
- Test section conditions including static and total pressure and total temperature.
- Model pitch and side slip angles during wind tunnel tests and ground height, pitch and roll angle during static ground tests.

5.0 TEST RESULTS

5.1 General Remarks

The results of the static hover and wind tunnel tests are presented in the following sections. In general the results are presented in terms of forces and moments which have had the direct thrust and ram forces removed. The resulting forces are given the subscript "A" (e.g., C_{LA}) for "aerodynamic". There are several reasons for presenting the data in this form as summarized below.

- 1.0 It is not always possible to exactly balance the two lift cruise fans. This can introduce rolling and yawing moments which must be removed in order to reveal the aerodynamic contributions.
- 2.0 The nose fan thrust was not exactly constant with angle of attack. This introduces a variation of pitching moment with angle of attack which is best removed to reveal the aerodynamic forces.
- 3.0 The ram force at a given C_j is not exactly representative of the airplane value. This occurs basically because of the mass flow added to drive the tip turbine and because of exhaust temperature differences between the model primary and the actual engine exhaust. Removing ram forces from the wind tunnel data facilitates correction to actual airplane ram force levels.

The relationship of the aerodynamic forces to the total model forces is illustrated in Figure 17 for a forward speed case representing a typical STOL liftoff condition. It is seen that the aerodynamic forces are small relative to the overall model forces. The aerodynamic data therefore represent the difference between two large numbers (total forces minus thrust and ram forces). This of course has an effect on the relative accuracy which can be achieved in the aerodynamic data.

In using the data to predict airplane performance the "thrust-drag"

bookkeeping system must be taken into account and it should be noted that there is a difference between the systems used for static data compared to that used for wind tunnel data. In using the static data the full scale thrust "out of ground effect" should be applied since the backpressure effect on thrust is already included in the data. However, in using the wind tunnel data any effects of backpressure on the full scale fan performance must be taken into account in the propulsion data used since the thrust removed from the wind tunnel data was based on actual measured backpressure. This subject is discussed in more detail in Section 4.4.

When using the "aerodynamic data" (C_{L_A} , C_{D_A} , etc.) contained herein to determine total forces on the configuration, the reverse of the wind tunnel data reduction procedure is applied. Unless identified otherwise, the aerodynamic data presented were determined with the distribution of thrust indicated in Figure 16. If this is the condition of interest the coefficients plotted can be used directly. If a different thrust distribution is being considered, the data concerning thrust distribution effects (e.g., Figure 31) can be used to determine an adjustment to the basic (e.g., Figure 30) data. The components of direct thrust and ram force for each of the three fans can then be added to the aerodynamic data to give total forces. In adding ram moments equations, C.11 through C.13 can be used. Typical measured ram forces from the wind tunnel data are summarized in Figure C.3. However, in predicting airplane forces it is recommended that ram forces more representative of the flight propulsion system be used.

5.2 Static Hover Characteristics

5.2.1 Effect of Ground Height for Level Model Attitude

The effect of the ground on the induced forces for the basic VTOL configuration ($\lambda = 90^\circ$) is shown in Figure 18 for zero pitch and roll angles. It should be noted that all of the data plotted represent repeat runs of the same configuration. This was treated as a control configuration to establish repeatability. It is seen that all four runs have the same shape while one of the four is displaced by about 2% from the others. This shift was apparently associated with measuring accuracy and probably is not unique to this run. In interpreting the subsequent plots, where multiple runs are not available, this accuracy limitation on level must be considered. Since curve shapes are apparently not much affected, the magnitude of the ground effect can be determined as an increment from the free air level measured on the particular run involved.

It is seen in Figure 18 that the induced forces on this configuration are small. Based on the three consistent runs the lift force is within about $\pm 2\%$ of the direct thrust component depending on ground height. The regions of positive lift are associated with the "fountain" which forms under the model as the three jets impinge on each other as they spread out on the ground plane.

It is seen that at the largest ground height tested ($h/\bar{c} = 5$) the induced lift is negative and equal to about 2% of total gross thrust. Subsequent tests run with the model inverted to give a more certain "free air" condition have confirmed this 2% negative induced lift. Since the fans were installed on the model during calibration and were run individually, this indicates that when all three fans are run simultaneously, the lift realized is 2% less than the sum of the individual fan contributions to lift.

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The free air region extends down to about $h/\bar{c} = 3.3$ ($h = 8.4$ meters for the full scale T-39 demonstration) below which the ground effect begins to produce a lift increase. The beneficial ground effect peaks out at about $h/\bar{c} = 2.3$ and then decreases gradually until at $h/\bar{c} = .8$, the lift is back to the free air value.

The induced "drag" (longitudinal force) and pitching moments are small. The negative slope of pitching moment vs. height is apparently associated with a rearward displacement of the point of impingement of the fountain on the model. This would be expected from geometric considerations since the nose fan is pointed 14° rearward. The induced pitching moments of Figure 18 could be trimmed by changes in nose fan thrust of $\pm 4\%$ (from the free air setting) combined with $\pm 2\%$ changes in L/C fan thrust to hold total thrust constant.

5.2.2 Effect of Angle of Attack

The effect of angle of attack on the induced forces is shown in Figure 19 for the case of $\lambda = 90^\circ$. For positive angles of 5° to 15° the favorable fountain effect found at 0° does not occur while at $\alpha = -5^\circ$ the magnitude of the favorable fountain effect has increased considerably. This effect is related to the location of the fountain as seen in the photos of oil flow on the ground shown in Figure 20. The projection of the model was drawn on the floor to help visualize the fountain location relative to the model. The dark line is an accumulation of oil at the stagnation point where the three jets meet. Figure 21 presents a tracing from oil flow photos at various angles of attack. It is seen that the fountain moves forward with angle of attack and at $\alpha = 5^\circ$ it passes ahead of the wing and therefore produces little upward force on the model. Oil on the lower model surface indicated that the ground oil pattern was in fact, indicative of the location of the fountain impingement on the model. However, these model oil patterns were very faint and distinct photographs were not obtained.

For the pitch range tested the greatest lift loss due to ground effect is about 3% of gross thrust relative to the free air lift level.

At $\alpha = 10^\circ$, and $\alpha = 15^\circ$ the induced drag changes noticeably with height. This implies that horizontal force trim will occur at combinations of α and λ somewhat different than those calculated from direct thrust considerations alone.

The effect of angle of attack at $\lambda = 80^\circ$ is shown in Figure 22. The effect is similar to that found at $\lambda = 90^\circ$. As the model is pitched up the fountain moves forward of the wing and its effect is largely lost. There is, of course, a shift in the angle of attack at which the fountain is directly under the wing compared to the $\lambda = 90^\circ$ case.

5.2.3 Effects of Lift/Cruise Fan Tilt Angle

The effects of tilting the lift/cruise fans upon the hover characteristics is presented in Figure 23. This effect is similar to that of changing pitch angle. Reducing the tilt angle to 30° , thereby causing the fountain to move aft, produced a lift increase similar to that occurring at a pitch angle of -5° . Increasing the nacelle tilt angle to 95° moves the fountain forward and thereby reduces lift as did an increase in pitch angle. Increasing the nacelle tilt angle to 95° also altered the pitching moment characteristics as did positive pitch angles. Namely, producing more negative pitching moments upon approaching the ground.

5.2.4 Effect of Roll Angle

Data for combinations of angle of attack and roll are presented in Figure 24 and 25. The effect of roll angles of 10° and greater is seen to be unfavorable on lift, especially for angles of attack greater than zero. At $\alpha = 10^\circ$ and $\phi = -10^\circ$ the lift reduction is about 4% of gross thrust relative to the free air lift for the same run. This is the most adverse model orientation found on this test.

The induced side force tends to resist airplane motion that would be caused by a bank angle, thus providing a stabilizing influence. The magnitude of this influence, however, is quite small. For example, at $\phi = -15^\circ$, the induced side force is about 2% of thrust. For combined pitch and roll, side force, which is most affected by pitch changes,

only varies 2% for a 10^0 change in pitch. Induced rolling and yawing moment ratios change .01 or less for a 10^0 pitch variation.

5.2.5 Pitch and Roll Control Capability in Hover

The maximum pitch and roll control power in hover are shown in Figure 26. This control capability is compared to the thrust induced aerodynamic pitching moments and rolling moments produced with $\alpha = 10^0$ and $\phi = -10^0$. In both cases, the available control power is substantially greater than the induced aerodynamic moments for that airplane attitude. The control power shown is that available when the total airplane thrust is held constant. The maximum nose up pitching moment is generated by increasing the full scale airplane nose fan thrust to maximum (2495 newtons, 11,150 lbs) while reducing the thrust of each L/C fan by half the nose fan increment. For each of the three fans, maximum thrust is determined by the fan blade pitch limit. For the Boeing 1041-135-2R configuration of a lift cruise fan V/STOL airplane (Reference 2), the nominal thrust of the three fans is:

L/C = 2018 newtons (8975 lbs)
Nose = 1877 newtons (8350 lbs)

Therefore, the L/C fans are reduced to 1709 newtons (7680 lbs) thrust each when the nose fan is increased to maximum thrust. For maximum nose down pitching moment, each L/C fan is increased to maximum thrust (2495 newtons) while reducing the nose fan thrust by an amount equal to the total increase in L/C fan thrust. The nose fan thrust is therefore reduced to 922 newtons (4100 lbs). For roll control, one L/C fan is increased in thrust (up to the maximum) while the other L/C fan is decreased by an equal amount

5.2.6 Static Test Result Summary

In summary, the propulsion induced aerodynamic effects upon the hover configuration of the T-39 lift/cruise fan airplane are small and show no sudden changes with ground height, pitch, or roll. In free air,

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there is a 2% loss of lift due to running all three fans simultaneously. The largest lift loss due to ground effect occurs at a combination of 10^0 of pitch and 10^0 of roll where lift is 4% of gross thrust below the free air value. From the induced aerodynamic moments observed during the static ground tests, the greatest variation in thrust of any one fan required for trim is 7%. This variation is required of the front fan for pitch trim and assumes that the thrust of the two lift/cruise fans is altered by 3.5% to maintain a constant level of total thrust. The variation is from a setting for trim in the static free air hover condition.

5.3 Jet Induced Aerodynamic Forces and Moments at Forward Speeds

The presence of the nacelle and the fan efflux produce large interference forces on the model. These forces are a strong function of the thrust coefficient and the angle and location of the jet relative to the wing. This is essentially a jet-flap type of effect with the added complication that the nacelle, at high λ 's represents a large bluff body. The data of the following sections are presented as a function of C_j , the total thrust coefficient of all three fans. Unless otherwise noted, the distribution of thrust among the three fans is as indicated by Figure 16. Some of the plots contain the notation " $C_j = \text{RAM}$ ". This indicates that the gross thrust was set to a value approximately equal to the ram drag by adjusting the RPM until fan exit total pressure was approximately equal to free stream total pressure.

5.3.1 Effect of Thrust on Aerodynamic Lift

The induced effect of the propulsion system on lift is illustrated in Figure 27 for a value of thrust coefficient representative of STO lift-off. The direct thrust component has been subtracted from the data and the baseline run shown is the nacelles-off case. It is seen that the propulsion system induces an increase in the lift coefficient at all angles of attack. The maximum aerodynamic lift coefficient and the angle at which it occurs are both increased by the propulsion system. For both nacelle tilt angles shown ($\lambda = 50^\circ$ and $\lambda = 90^\circ$) the increase in maximum aerodynamic lift coefficient is about .28 which is a 19% increase from the nacelles-off value.

The propulsion induced lift effects as a function of thrust coefficient are summarized in the following paragraphs.

5.3.2 Effect of Thrust on Aerodynamic Lift at Constant Angle of Attack

The effect of thrust coefficient on model aerodynamic lift at various λ 's is summarized in Figure 28. For $\lambda = 0^\circ$ the effect of C_j is seen to be small. It is slightly favorable when only the L/C fans are

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operating and slightly adverse when, in addition to the L/C fans, the nose fan is operated at about 30% thrust (the nose fan doors were open in both cases).

When λ is increased to 50° the lift at $C_J = 0$ is unaffected (although this may result from a trade of lift gain on the nacelles for lift loss on the wing). As C_J is increased a positive lift increment of about .4 is realized followed by a declining lift above $C_J = 3.0$. The decrease at higher C_J 's is apparently associated with the adverse effect of the nose fan thrust as will be discussed below.

At $\lambda = 90^\circ$ and $C_J = 0$ the nacelle causes a C_{LA} loss of about .4 relative to the level at $\lambda = 0^\circ$, $C_J = 0$. The nacelle is very close to the wing trailing edge and its pressure field apparently prevents development of the normal flow over the inboard wing. As C_J is increased the C_{LA} increases and at $C_J = 5.5$ it reaches a level about .6 above the $\lambda = 0^\circ$ level. At this point, the curve has become very flat and possibly would turn down at higher C_J due to the adverse nose fan effect. The beneficial effect of C_J extends to higher values at $\lambda = 90^\circ$ (compared to $\lambda = 50^\circ$) apparently because of the relatively greater influence of the L/C jets at the higher angle.

The adverse nose fan effect mentioned above is demonstrated in Figure 29. In this case, the incremental induced lift due to nose fan thrust is plotted with the base level being the value of induced lift when the nose fan is set for static moment trim. The nose fan thrust values which result in static moment trim at each λ are indicated on the plot. For the STOL case ($\lambda = 50^\circ$) it is seen that reducing the nose fan thrust to zero produces a C_{LA} gain of .3. The effect is similar at $\lambda = 0^\circ$ (note that the nose fan thrust used as reference level was lower at $\lambda = 0^\circ$). At $\lambda = 90^\circ$ the range of nose fan thrusts tested was too small for good definition but a similar trend is indicated within the data accuracy.

It is of interest to compare the present lift results with those of

Reference 1 which represent a large scale model of a somewhat similar configuration. Both configurations have a nose fan and two lift/cruise fans, however, the present configuration obtains thrust vectoring by tilting the L/C nacelles while the Reference 1 configuration utilizes a thrust vectoring exhaust nozzle. In order to make the comparison the data of the present test were converted to the form found in Reference 1. This involves taking the difference in aerodynamic lift between a powered lift case and a reference case and dividing by the total gross thrust of the former. The reference case used, as in Reference 1, is with nose fan covered and L/C thrust undeflected at a large V/VJ value. The results are shown in Figure 30. It is seen that the propulsion induced lift of the two configurations is similar. The present configuration has somewhat higher propulsion induced lift at typical STO jet deflection angles.

5.3.3 Effect of Thrust on Maximum Aerodynamic Lift

The effect of thrust on maximum aerodynamic lift coefficient is shown in Figure 31. At $\lambda = 0^\circ$ the effect of thrust is slightly beneficial when the nose fan is at zero thrust but is detrimental when the nose fan is set at a value which trims the static thrust moments. Increasing λ progressively lowers $(C_{L_A})_{\max}$ when the thrust is zero. As the thrust is increased at $\lambda = 50^\circ$ or $\lambda = 90^\circ$ a further decrease in $(C_{L_A})_{\max}$ occurs up to about $C_J = 1.5$ followed by a recovery of maximum lift at higher C_J values. At the highest C_J values tested the maximum aerodynamic lift is increased by about .2 compared to the $\lambda = 0^\circ$, $C_J = 0$ case.

The effect of thrust balance between the L/C fans and the nose fan on the maximum aerodynamic lift is shown in Figure 32. At $\lambda = 0^\circ$ and $\lambda = 50^\circ$ the nose fan thrust is clearly detrimental to $(C_{L_A})_{\max}$ while at $\lambda = 90^\circ$ the range of nose fan thrust values tested was too small to reveal any effect on $(C_{L_A})_{\max}$.

5.3.4 Effect of Thrust on Aerodynamic Drag

The effect of thrust on the aerodynamic drag is illustrated in Figure 33. In this case, the data have been plotted in terms of drag increment from the zero thrust case. This presentation is used because it is felt that the actual drag levels of the model must be adjusted for the nonrepresentative strut fairing as discussed later on (see configuration component effects).

Figure 33 shows that the effect of power is to increase the aerodynamic drag at a given value of aerodynamic lift. As C_L is increased, a limiting value of this adverse drag increment is reached with both the limiting drag increment and the C_L at which it is reached being dependent on λ .

The effect of thrust balance between the nose fans and the L/C fans is shown in Figure 34. Compared to the very large effect of thrust level, it is seen that the thrust distribution has a relatively small effect on drag. For the CTOL case ($\lambda = 0^\circ$) increasing the nose fan thrust ratio from 0 to .3 has a beneficial effect while the STOL case ($\lambda = 50^\circ$) shows an adverse effect of increasing nose fan thrust. The range of data at $\lambda = 90^\circ$ is too small to reveal any trends.

5.3.5 Effect of Thrust on Pitching Moment

The induced effect of thrust on the aerodynamic component of pitching moment is shown in Figures 35 thru 37 for tail-on and tail-off cases at various engine tilt angles. Some general observations will be made from these curves before summarizing the effects of power upon trim requirements and stability parameters in the succeeding paragraphs.

When the nacelles are tilted (Figures 35 and 36) the pitching moment curves are seen to display a double break characteristic with a first nose-up break occurring at a lift coefficient between about .6 and 1.2, depending on nacelle tilt angle and thrust level, and a second break occurring at higher lift coefficient when the wing stalls. With the

nacelles not tilted (Figure 37), the break is not so prominent. In the following discussion reference to low C_L and high C_L will be used to indicate the regions which are, respectively, below and above the first pitching moment break.

At $\lambda = 90^\circ$ (Figure 35) it is seen that the tail provides a large increase in stability at low C_L values but not at high C_L 's. The tail contribution to stability at low C_L 's is seen to increase with increasing thrust coefficient.

At $\lambda = 50^\circ$ (Figure 36) the tail-off configuration is seen to be stable at low C_L values. However, the tail contribution to stability in this case is much less than at $\lambda = 90^\circ$. Also the sensitivity of tail-on stability to power at low C_L 's is greatly reduced at $\lambda = 50^\circ$ compared to $\lambda = 90^\circ$. At higher C_L 's the tail is seen to provide a stability increase at the high C_J values.

In the loiter configuration, flaps up with $\lambda = 0^\circ$, an increase in thrust coefficient produces a small increase in tail-off stability as seen in Figure 37. It is also seen that when the tail is on, an increase in C_J appears to have little effect on stability, as far as can be determined with the relatively small number of data points for the power-off case. This data represents the high speed end of the V/STOL flight envelope. For this case, it is important that the tail provide all of the stability and control functions. It is clear that the tail is providing effective alpha stability. Trim requirements appear to be within airplane capability with the nose fan disengaged.

5.3.6 Effect of Thrust on Trim Requirements

Because of the high degree of non-linearity in the data (Figures 35 and 36) any summary of trim requirements becomes dependent on the conditions chosen to perform the analysis. To illustrate the effect of power and nacelle tilt angle on trim requirements, pitching moment will be summarized for a constant value of aerodynamic lift coefficient, $C_{L_A} = 1.0$.

The effect of thrust on pitching moment for the tail off configuration is summarized in Figure 38. The result obtained for aerodynamic pitching moment is strongly dependent on the manner in which ram moments are accounted for. The method of removing ram moments has been described in detail in Appendix C. The effect of ram drag is to produce a large positive pitching moment proportional to C_j . It is seen in Figure 38, that for all values of λ this moment is at least partially compensated by the negative value of aerodynamic pitching moment induced by thrust effects. For C_j below 3.0 at $\lambda = 50^\circ$ the negative induced moment is even greater than the positive ram moments with the result that the total moment is negative. For $\lambda = 0^\circ$ and nose fan inoperative, the total moments are nose down. By first removing the direct thrust moments, then removing the ram moments, it can be seen as stated above, that for this case also, the ram moments are positive and are partially offset by the negative aerodynamic moments. As discussed in Appendix C, the model ram force is less than that which will occur full scale at the same thrust coefficient. The required correction to full scale has been made for the cases shown in Figure 38, as shown by the curves thus labeled in that figure.

The moments resulting from ram forces and aerodynamic forces can be trimmed by varying the thrust distribution, by changing horizontal tail incidence, or by some combination of these two methods. Figure 39 illustrates the change in thrust distribution required if the moments are trimmed entirely by means of thrust forces (the data shown are tail-off). An iteration was performed to allow for the fact that ram moment changes as the thrust distribution is changed. The ram moments used were those which had been corrected to full scale and it was assumed that the nose fan and L/C fan thrusts were collectively changed so as to hold total thrust constant. It is seen that the nose fan thrust ratio reduction required for trim is about .10 at $\lambda = 50^\circ$ and .20 at $\lambda = 90^\circ$ relative to the static hover case.

5.3.7 Effect of Thrust Upon Longitudinal Stability and Control Parameters

Several longitudinal stability and control parameters have been evaluated

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for the tested configurations and are summarized in Figures 40 thru 44. These include lift curve slope, neutral point location, horizontal tail effectiveness and downwash at the horizontal tail. It should be remembered that only the aerodynamic characteristics are being analyzed in that the effects of thrust and ram forces have been excluded.

The aerodynamic lift curve slopes are presented in Figure 40. The nacelles are shown to produce an increase in lift curve slope while in their untilted position. Tilting the nacelles degrades the lift curve slope. The effect of thrust is to increase lift curve slopes for the CTOL and STOL configurations, while it was found that thrust increases reduce the lift curve slopes of the cruise and VTOL configuration.

As discussed in the previous section, the pitching moment curves exhibit a double break characteristic. Hence, evaluation of the aerodynamic neutral point location (i.e., the center of gravity position at which the aerodynamic contribution to static stability is zero) was conducted separately for low lift values (Figure 41) occurring below the first break in the pitching moment curve and for higher lift values (Figure 42) occurring between the first break and wing stall. The analysis was conducted only at low lift values for the VTOL configuration since the first break occurs at fairly high lift levels. Neutral point locations are presented with the horizontal tail set at zero incidence and with tail-off. For comparison, the neutral point location with nacelles and horizontal tail removed is also presented. At the lower lift levels (Figure 41) it is seen that for all tilt angles the complete configuration is more stable than the nacelles-off tail-off configuration. In all cases presented, the horizontal tail contributes to the stability. The smallest contribution made by the tail is at $\lambda = 50^\circ$. The stability of the cruise and STOL configurations is relatively insensitive to thrust increases, whereas the stability of the VTOL configuration is significantly increased at high thrust values. For high lift levels (Figure 42) the general trend with thrust for the cruise configuration remains the same. For $\lambda = 50^\circ$ the complete configuration is slightly less stable than the nacelles-off tail-off configuration at low C_j .

values and shows a further decrease in stability at thrust coefficients above 3.5. Comparison of tail-on versus tail-off for both the cruise and STOL configurations indicates that the tail still contributes to the longitudinal stability.

Figure 43 presents the horizontal tail control effectiveness. The control effectiveness is the change in pitching moment due to changes in horizontal tail incidence ($\partial C_{PM} / \partial \alpha$). The figure compares the tail control effectiveness measured in the test with a pre-test theoretical estimate. For the cruise and STOL configurations, the tail effectiveness is near the theoretical value and is not appreciably affected by thrust variation. The tail effectiveness of the VTOL configuration is far below the theoretical level when power is off, but is shown to approach the theoretical level with increasing thrust.

The downwash gradient at the horizontal tail is shown in Figure 44. This parameter was computed from the intercepts of the lower part of the plots of aerodynamic pitching moment as a function of angle of attack for horizontal tail-off and for horizontal tail-on at two different incidence angles. The intercepts of the tail-off and tail-on curves provide the angle of attack at which downwash is equal to the horizontal tail incidence. For the cruise and STOL configurations, an increase in thrust coefficient increases the downwash gradient while it decreases with thrust for the VTOL configuration. This relates to the previous discussion of neutral point location which showed that the contribution of the horizontal tends to diminish with thrust for the STOL configuration and tends to increase with thrust for the VTOL configuration.

5.4 Cross-Wind and Differential Thrust Effects

The effects of differential thrust and sideslip upon the model in its VTOL configuration ($\lambda = 90^\circ$) were investigated at sideslip angles of 0° , 5° and 10° (Figures A.7, A.8, and A.9 in Appendix A). Yawing moments are somewhat difficult to interpret due to large zero shifts in that balance component. This problem is illustrated in Figure 45 which shows yawing moments as a function of angle of attack for several configurations and sideslip angles with $\lambda = 90^\circ$ and $C_J = 3.7$. It is seen that the data at zero sideslip indicates a yawing moment coefficient of about $+0.03$. However, the zero shifts encountered for the group of runs shown in the figure ranged from 0 to $+0.02$ with a shift of $+0.015$ being typical. This amount of uncertainty could well affect the interpretation of the data. Because of this problem, it is felt that the cross-wind characteristics are not definitive. Assuming, however, that the model had no residual yawing moments at $\beta = 0^\circ$, the data at both sideslip angles would indicate the configuration was stable in that mode over most of the angle of attack range with neutral stability at low angles of attack. The contribution of the vertical tail surface is marginally stabilizing at $\beta = 5^\circ$ and destabilizing at $\beta = 10^\circ$.

The lateral-directional static derivatives for the VTOL configuration at angles of attack near zero are shown on Figure 46. This analysis indicates that for the most part, the configuration is statically stable except for the directional stability. The directional stability is degraded by increased thrust but improves dramatically with angle of attack as shown on Figure 46 by the data at $\alpha = 8^\circ$. As mentioned above, if the model is assumed to have no residual yawing moments at $\beta = 0^\circ$, the directional stability would be neutral to positive for all thrust values. This analysis is based on the data shown on Figure A.10 thru A.17 of Appendix A. The increments shown are the difference between the $\beta = 0^\circ$ and $\beta = 5^\circ$ data.

The effect of differential thrust upon the model in its VTOL configuration was found to be negligible (Figures A.7, A.8 and A.9 in Appendix A). The

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thrust differential amounted to as much as a 20% increase in the nose fan thrust from its nominal setting for the fore-and-aft differential, and a 20% reduction in one of the lift/cruise fans for the side-to-side differential. In some cases, the differential was produced by increasing the thrust of one lift/cruise fan while reducing the thrust of the other. The differential thrust effects did not change with side-slip angle. Note that in Figures A.8 and A.9, the differential thrust effects were only explored near zero pitch angle.

5.5 Configuration Component Effects

A limited amount of configuration buildup testing was performed as summarized in this section. The discussion will generally be limited to the lift and pitching moment effects because the drag data are generally not sufficiently accurate to allow comparisons of relatively small differences between configurations. Some of the drag data are discussed in a separate paragraph at the end of the section.

5.5.1 Effect of Nacelles

The effect of nacelles on the tail-off lift and pitching moment data is shown in Figure 47 and Figure 48 for the CTOL and loiter configurations, respectively. It is seen that the nacelles produce a negative change in lift at low angles of attack but increase the lift curve slope and result in an increase of about .13 in maximum lift coefficient. For both flaps-up and flaps-down cases, the nacelles produce an increase in tail-off stability.

5.5.2 Effect of Flaps

The effect of extending the trailing edge flaps and leading edge slats is shown in Figure A.33 (Appendix A) for the CTOL configuration and Figure A.37 (Appendix A) for the nacelles-off case. For both the nacelles-off case and the CTOL at ram thrust case, the increment in maximum lift coefficient due to flaps is about .50. The somewhat larger increment of .65 for the power-on case is due to a premature stall for the power-on flaps-up case. For the CTOL power-off and the nacelles-off cases, there is no change in stability due to flaps while the flaps are slightly stabilizing for the CTOL power-on case. It is noted that the negative shift in pitching moment due to flaps is much less with the nacelles on than for the nacelles-off case.

5.5.3 Effect of Aileron Droop

The effect of drooping both ailerons 15° on the STOL configuration is shown in Figure A.28 (Appendix A). It is seen that a gain in maximum

lift coefficient of about .2 was realized.

5.5.4 Effect of Nacelles on Drag Level

The effect of the nacelles upon drag for the ram thrust case is summarized in Figure 49 in terms of C_{DA} as a function of nacelle tilt angle. Two curves are shown, one for the basic nacelle strut and one for a modified nacelle strut. Out of necessity, a portion of the lift/cruise fan nacelle strut was rotated with the nacelles during the wind tunnel test program to cover plumbing to the turbopowered simulators and the "blown" primary. That fairing, which would not be present on the full scale airplane, had a large effect upon the drags of the STOL and VTOL test configurations. To assess the drag difference due to the fairing, a test run was made with the fairing removed and tape and wax applied to cover and fair over the exposed plumbing and instrumentation. The basic fairing and an indication of the wax fairing are shown in Figure 7. The reduction in drag coefficient due to removing the basic fairing amounted to about .15. A further reduction is anticipated if the plumbing and instrumentation and their fairing could have been eliminated. Also presented in Figure 49 is an estimate of the STOL configuration drag from Reference 9. It is seen that the estimated level is very close to the level measured on the model. This indicates that the estimate is probably conservative for the power-off case since the model drag level is believed to be subject to further reduction with a more representative nacelle strut fairing.

6.0 CONCLUSIONS

1. Propulsion induced forces and moments measured during static ground tests were found to be small and showed no sudden changes with variation of ground height, pitch and roll.
2. A hover lift loss of 2% of gross thrust was found for heights far from the ground. With the model in a level attitude, there was no additional lift loss due to ground effect at any ground height. The largest induced lift loss due to ground effect occurred at a combination of 10^0 of pitch and 10^0 of roll where a loss equal to 5% of gross thrust was measured.
3. Observed induced moments for static hover in ground effect can be trimmed with less than 7% variation of nose fan thrust from the value for free air static trim.
4. Usable propulsion induced lift for the STOL configuration was found to be 19% of the nacelles-off lift level. The increment is based on maximum lift.
5. Drag levels measured with lift/cruise fans tilted are not representative of the drag of the full scale airplane due to non-representative nacelle struts which rotated with the lift/cruise fans out of necessity to cover plumbing to the turbo-powered simulators. The aerodynamic drag tended to increase with thrust.
6. Propulsion induced aerodynamic (wing) pitching moments were large nose-down moments with the lift/cruise fans tilted. The ram moments are counteracting, thereby reducing trim requirements.
7. The effect of the nacelles on longitudinal stability is very dependent on nacelle tilt angle, angle of attack, and thrust coefficient. In general, the nacelles tend to be stabilizing at low nacelle angles and destabilizing at the highest nacelle angles.

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8. The effect of increasing thrust coefficient is generally to increase the longitudinal stability, especially at high nacelle tilt angles.
9. The VTOL configuration has good directional stability at high angles of attack which degrades as angle of attack is reduced.

7.0 REFERENCES

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11. Heyson, H. H., "Linearized Theory of Wind - Tunnel Jet Boundary Corrections and Ground Effects", NASA TR R-124, 1962.
12. Trussell, D. R., "Report on WT1533, A Wind Tunnel Data Reduction Program Used to Compute Wall Corrections Based on Heyson's Interference Theory", Boeing Document D6-40794TN, 1973.
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FIGURE 1 - VIEW OF MODEL MOUNTED IN STATIC TEST AREA

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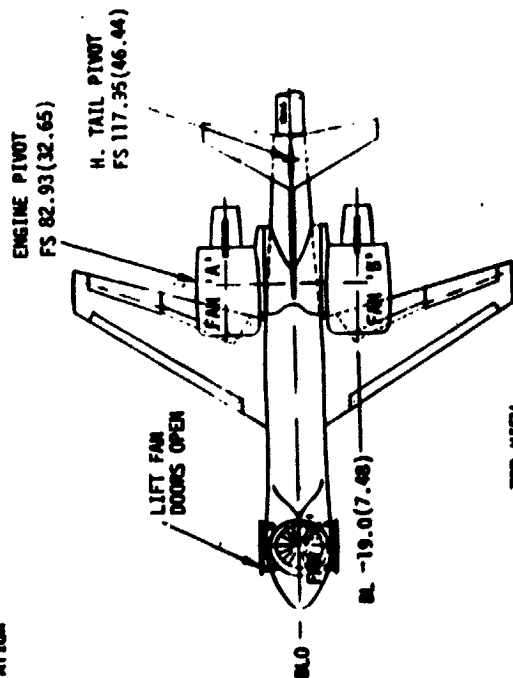
	WING	H. TAIL	V. TAIL
REFERENCE AREA $cm^2 (in^2)$	2000(435.17)	523 (81.07)	341(52.86)
SPAN CM (IN)	127.3(50.11)	51.26(20.18)	23.32(9.18)
MAC CM (IN)	30.82(9.46)	10.24(4.03)	14.62(5.76)
ASPECT RATIO	5.77	4.992	1.59
TAPER RATIO	.325	.35	.367
WEEP, 25% LINE	28.95°	30°	30°

REF. ML 23.88 IS T-39 REFERENCE WATER LINE (FULL SCALE ML 100)

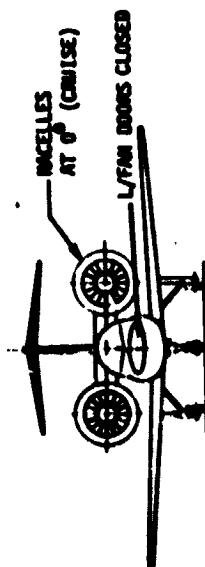
BL = BUTT LINE

FS = FUSELAGE STATION

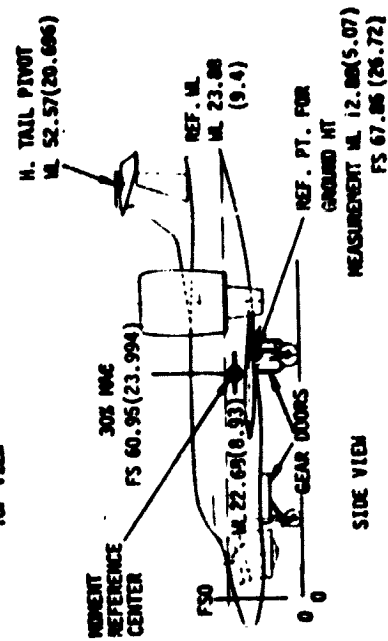
ML = WATER LINE



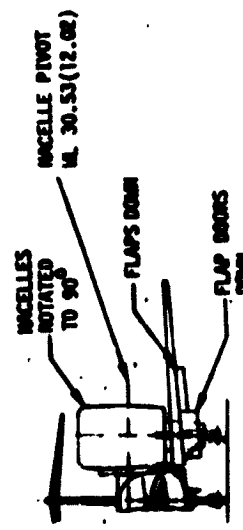
TOP VIEW



FRONT VIEW



SIDE VIEW



VTOL CONFIG
FRONT VIEW

DIMENSIONS: CM (IN) MODEL SCALE = .084

FIGURE 2 MODIFIED T-39 LIFT/CRUISE FAN AIRPLANE MODEL

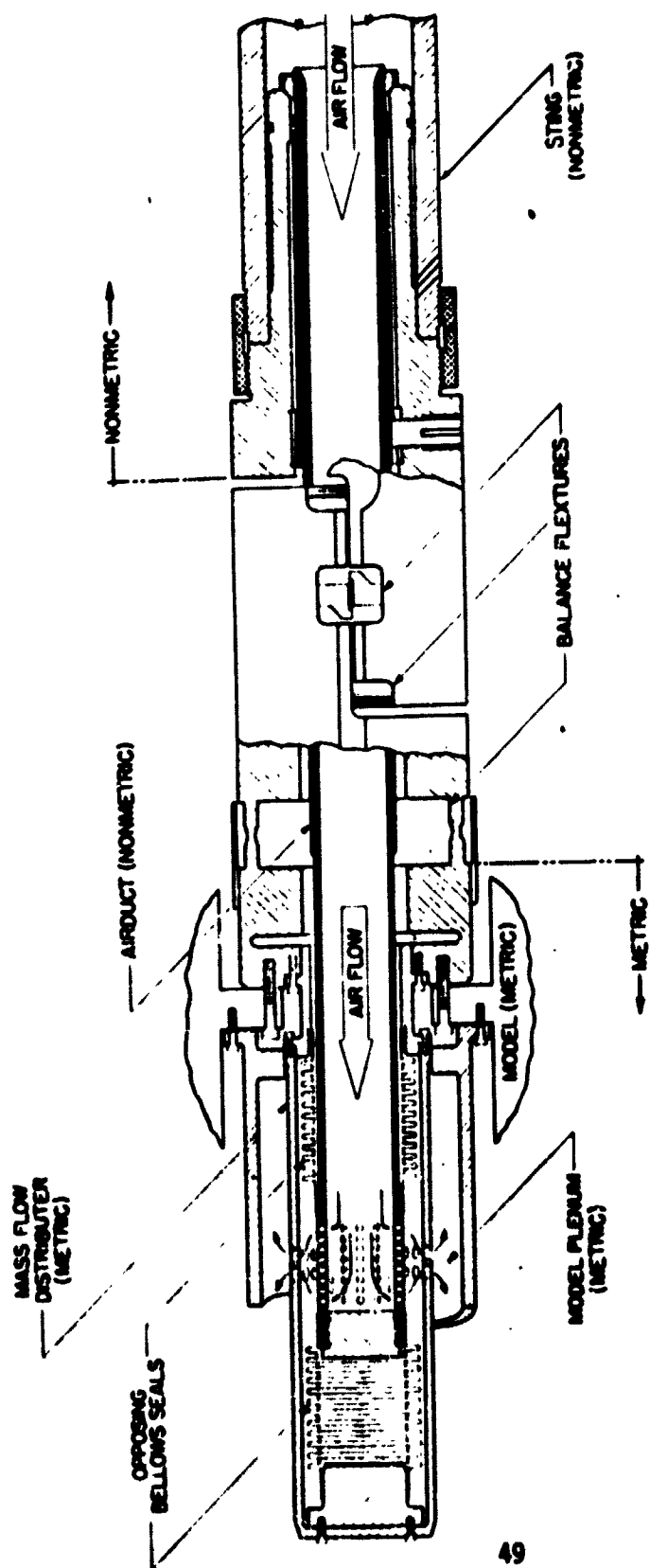


FIGURE 3 SCHEMATIC OF "FLOW-THRU" INTERNAL BALANCE



FIGURE 4 MODEL 1D-457 TIP DRIVEN FAN AND DRIVE PLENUM

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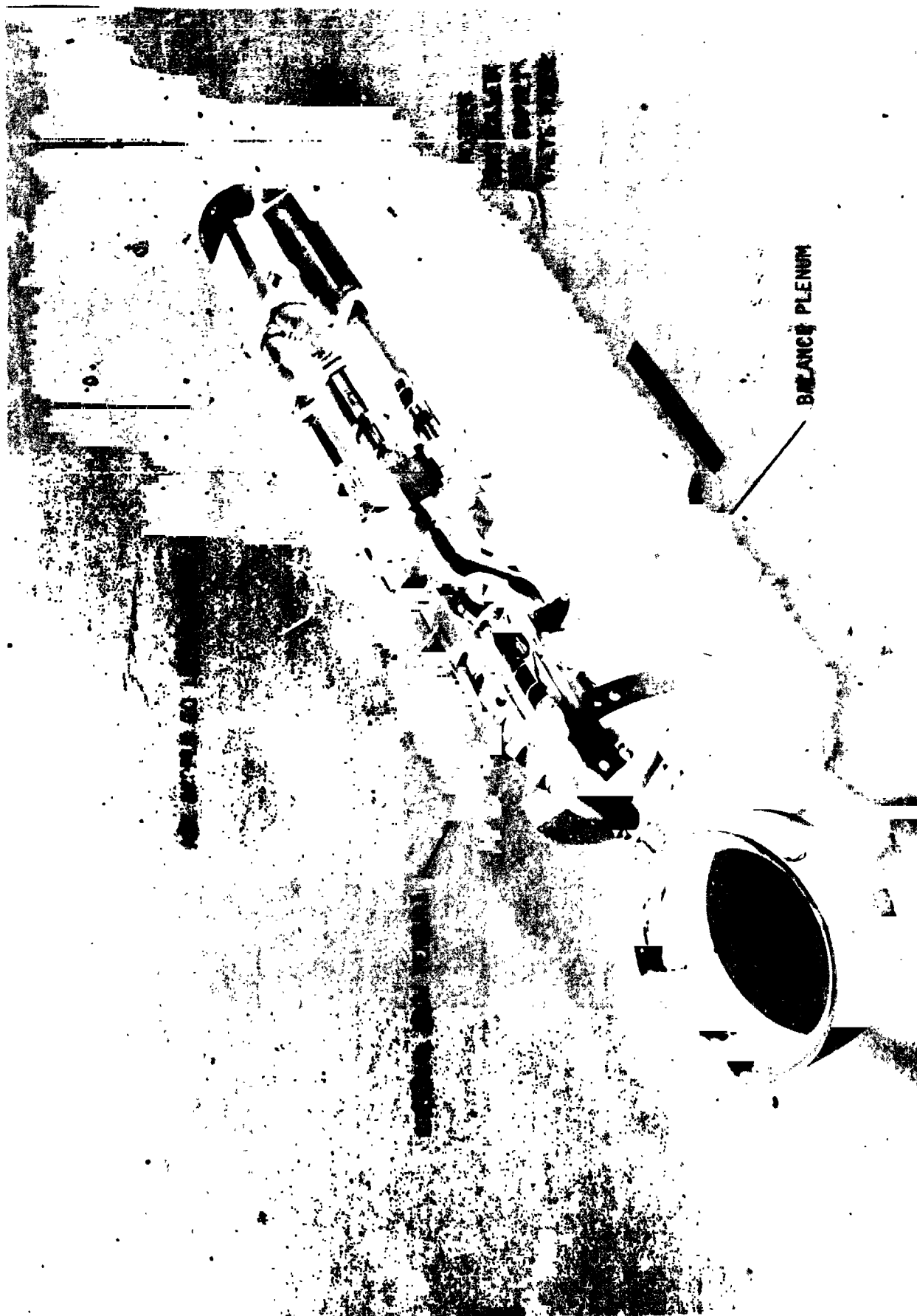


FIGURE 5 MODEL BASIC STRUCTURE

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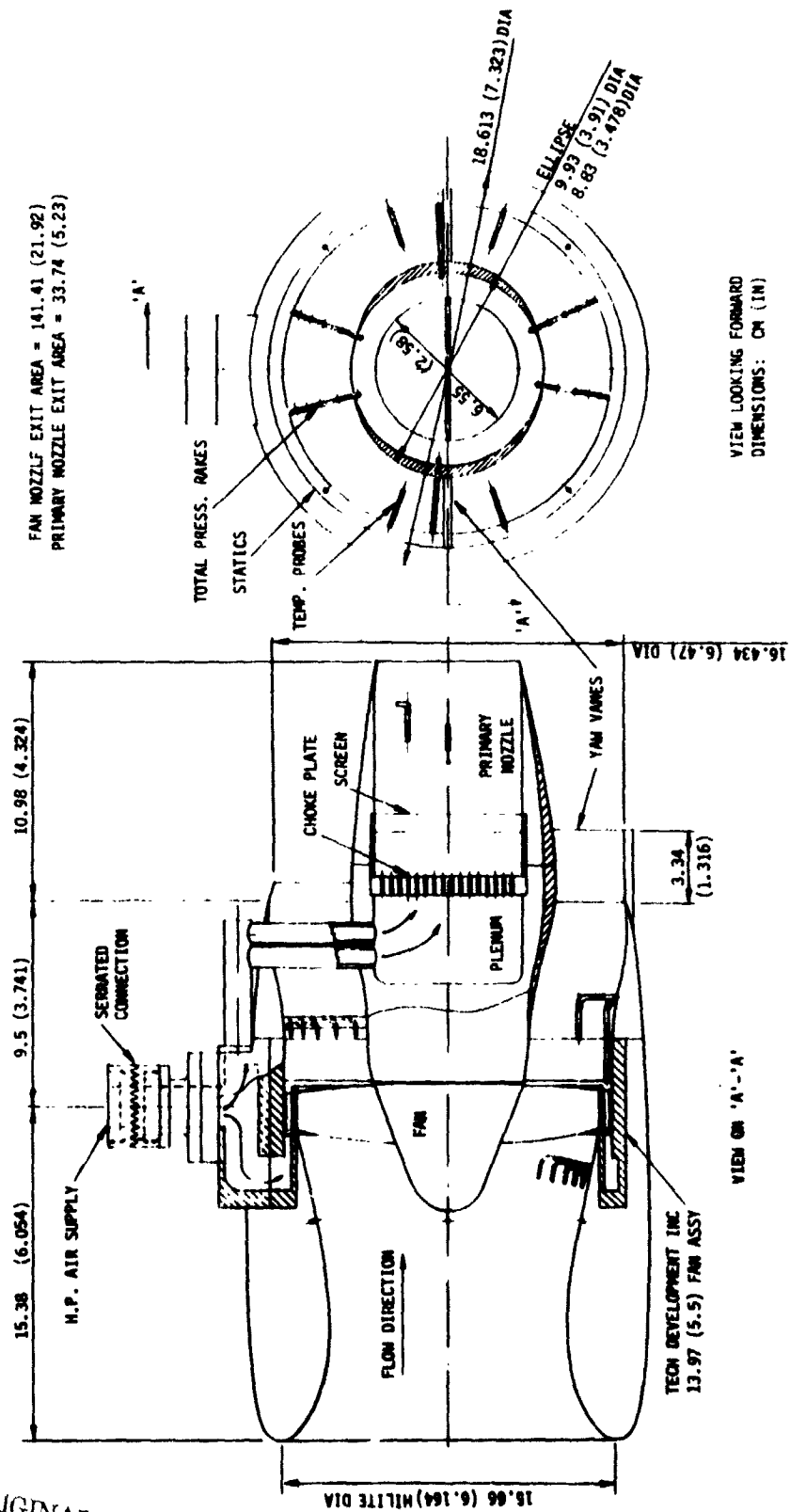


FIGURE 6 ASSEMBLY OF L/C FAN NACELLE

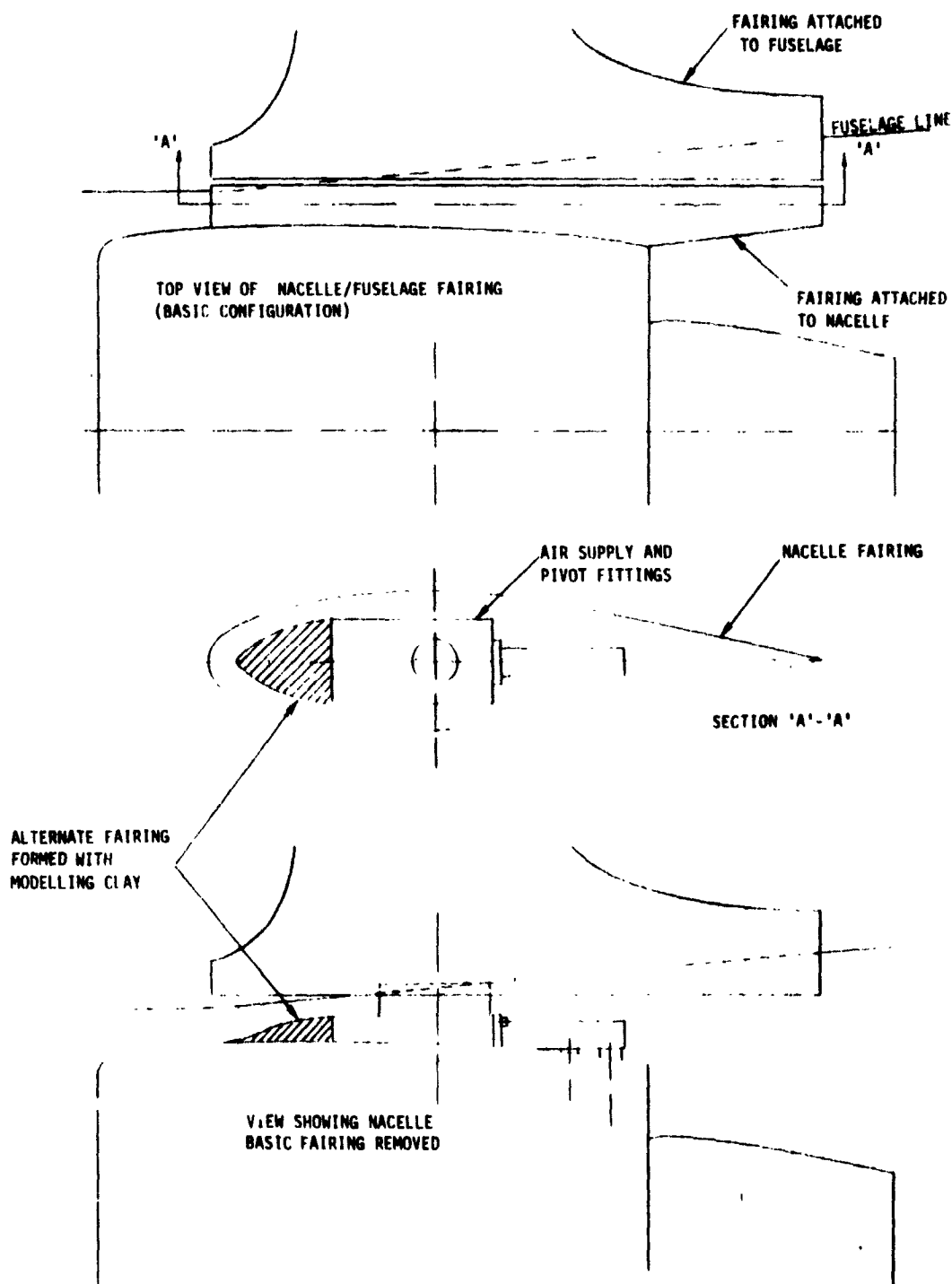
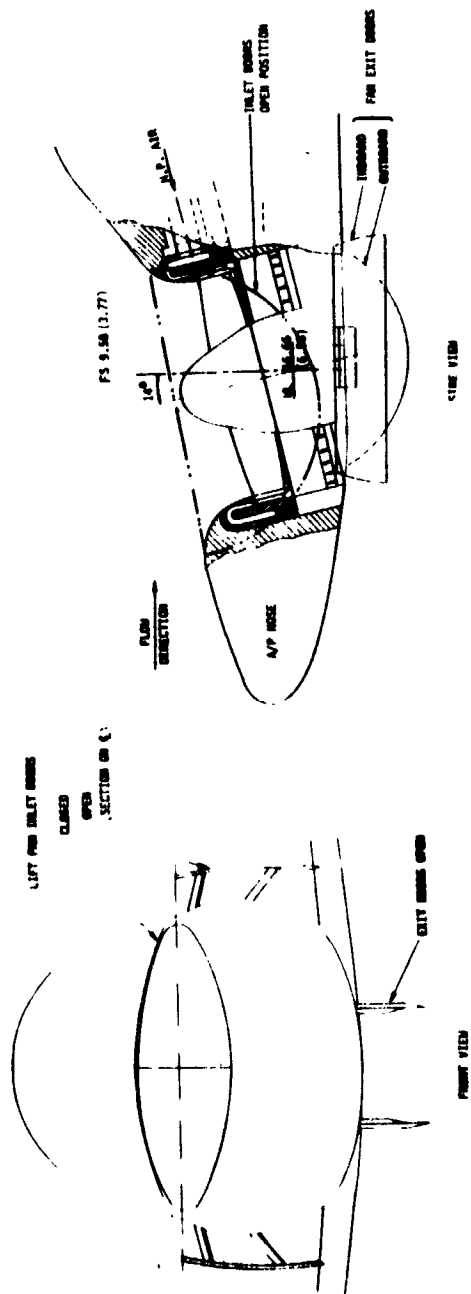


FIGURE 7 DETAILS OF NACELLE/FUSELAGE FAIRING

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DIMENSIONS: IN (MM)

FIGURE 8 ASSEMBLY OF LIFT FAN

WING

-
- WING
- L/E SLAT
- FLAP
- FLAP DOOR
- AILERON
- HORIZ. TAIL
- VERT. TAIL
- SECTION: NACA 0015
TIP NACA 0010
- SECTION: NACA 0010
- Dimensions (inches in parentheses):
- 33.31 (13.11)
 - 15.725 (6.191)
 - 12.14 (4.78)
 - 0.28
 - 38.62 (15.204)
 - 59.46 (23.408)
 - 63.74 (25.096)
 - 40.31 (15.87)
 - 51.12 (20.124)
 - 59.46 (23.408)
 - 33.013 (12.997)
 - 22.97 (9.043)
 - 19.82 (7.802)
 - 23.32 (9.18)
 - 1.79 (.705)
 - 24.07 (9.475)
 - FS 101.95 (40.138)
 - 30.8 (12.126)
 - 60°
 - 9.2 (3.62)
 - 15.19 (5.98)
 - 14.94 (5.88)
 - 20.26 (7.98)
 - 25.63 (10.09)

FIGURE 9 MODEL PLANFORM DETAILS

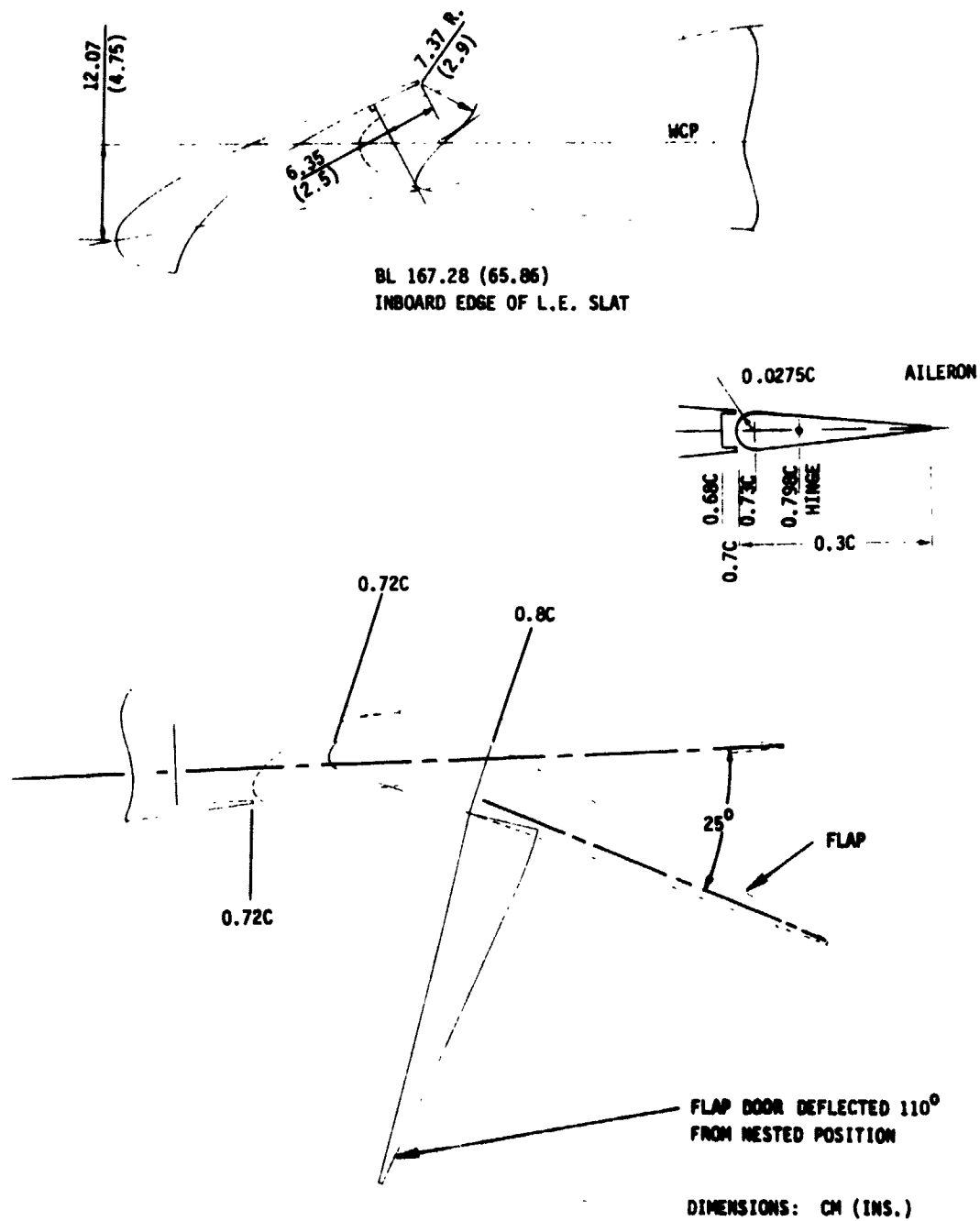


FIGURE 10 MODEL SECTION DETAILS

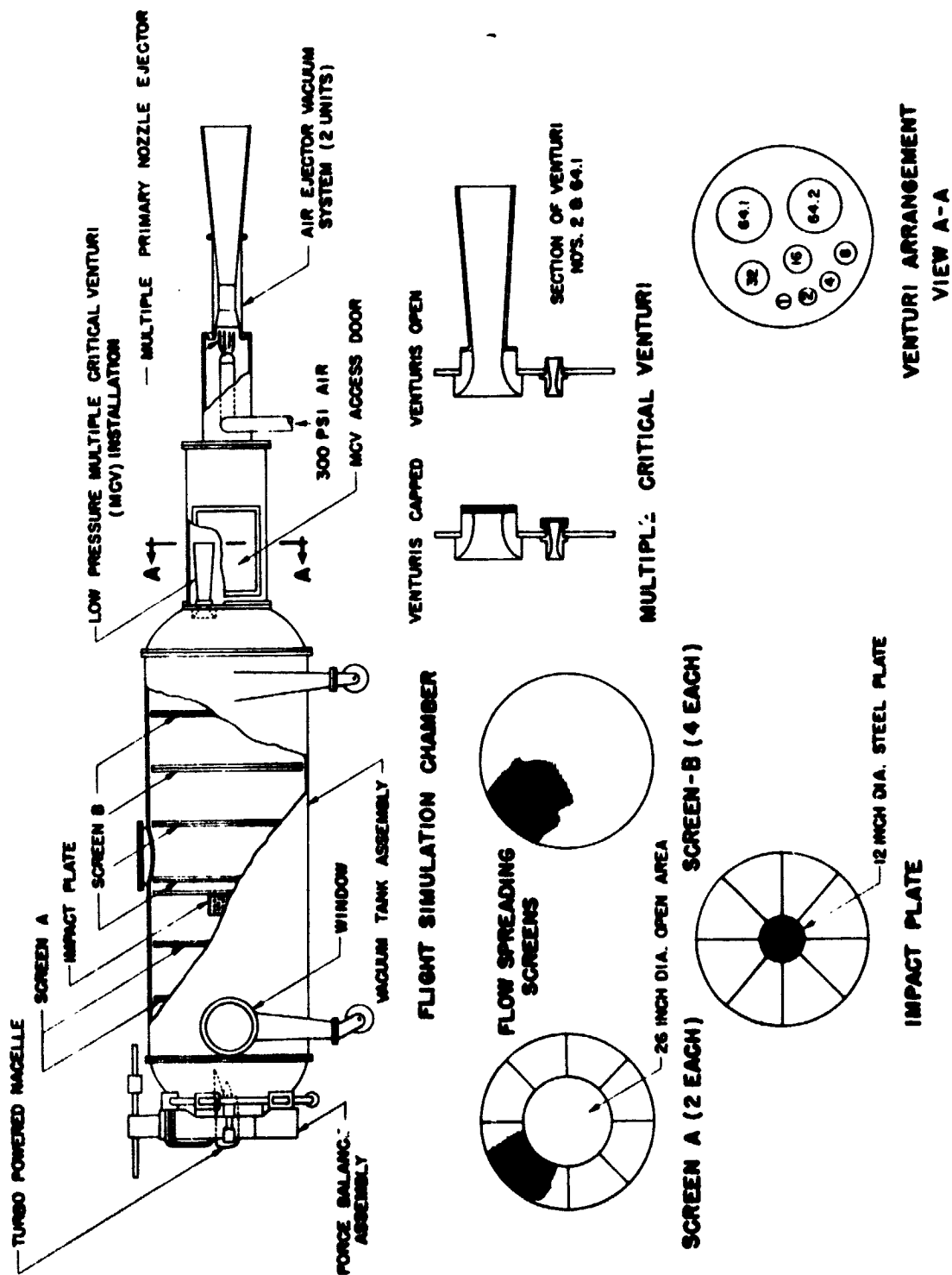


FIGURE 11 FLIGHT SIMULATION CHAMBER

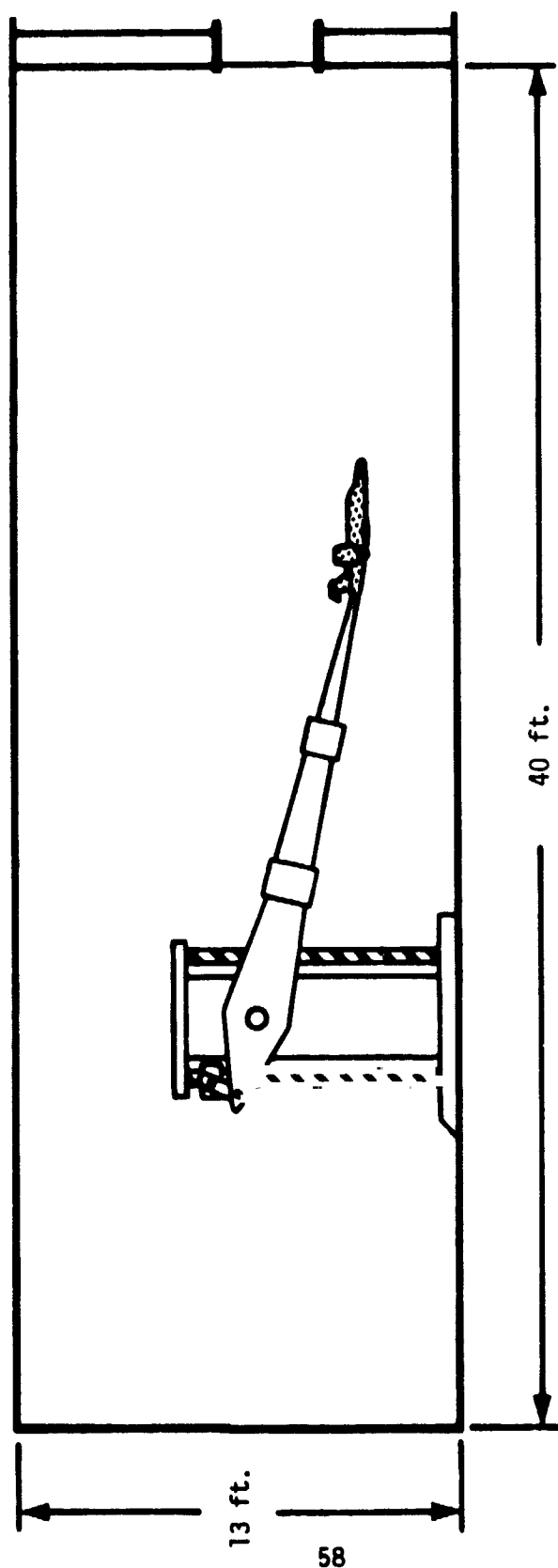


FIGURE 12 BOEING STATIC CHECKOUT FACILITIES

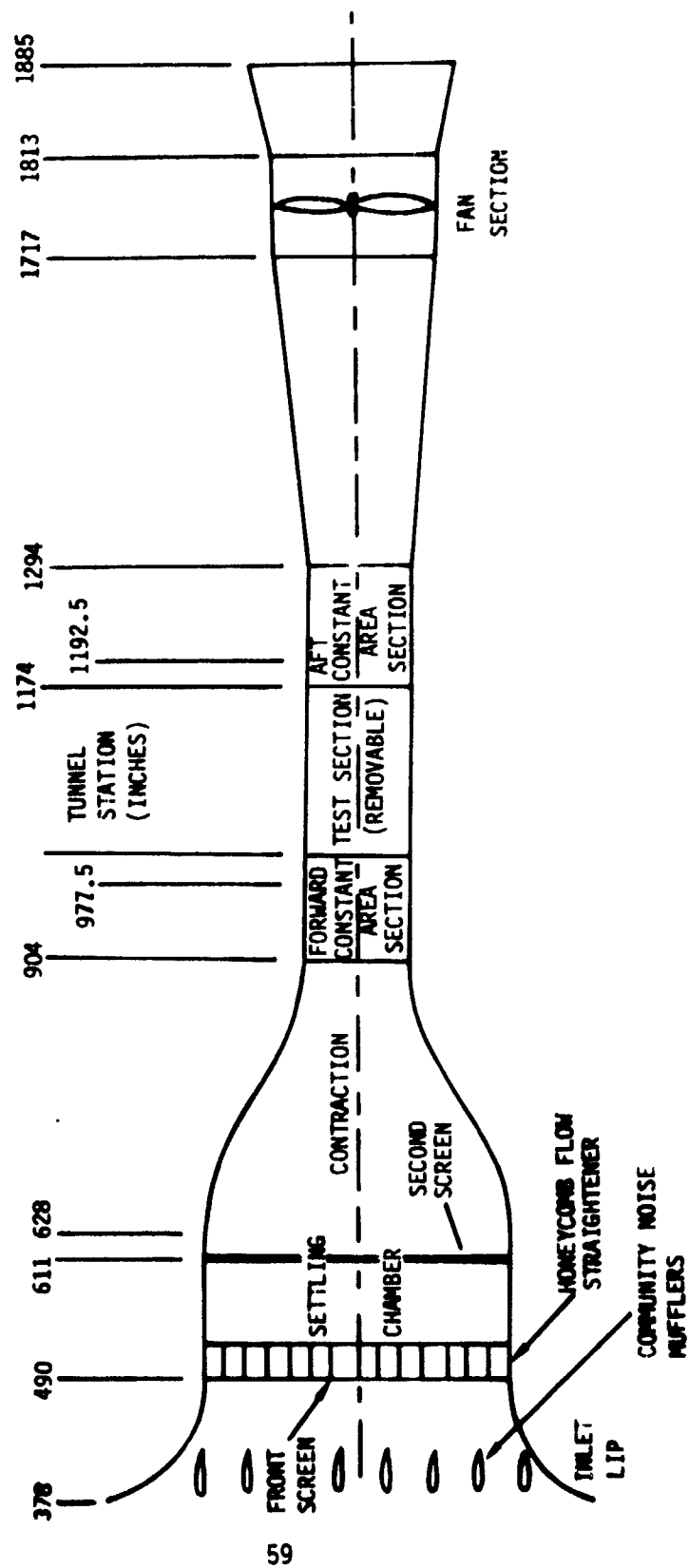


FIGURE 13 GENERAL ARRANGEMENT OF BOEING 9' x 9' WIND TUNNEL

PERREL CILING



FIGURE 14 MODEL INSTALLED IN THE BOEING 9' LOW SPEED WIND TUNNEL

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CONFIGURATION	VARIABLES EXPLORED						
	L/C FAN TILT ANGLE	PITCH ANGLE	YAW ANGLE	TUNNEL SPEEDS	THRUST COEF.	STABILATOR ANGLE	COMPONENT EFFECT
VTOL	80°, 90°, 95°	-8° TO + 30°	5°, 10°	65 KTS TO 120 KTS	0.0 TO 5.5	0°, 20°	STABILATOR, FIN, NACELLES, GEAR
STO	50°	-8° TO + 30°	—	65 KTS TO 120 KTS	0.0 TO 5.5	0°, 10°	STABILATOR, GEAR, AILERON DROOP (15°), NACELLES
CTOL	0°	-8° TO + 30°	—	120 KNOTS	0.0 TO 1.2	—	GEAR, NACELLES
LOITER	0°	-8° TO + 30°	—	120 KNOTS	0.0 TO 1.2	0°, 10°	STABILATOR, NACELLES

FIGURE 15 WING TUNNEL TEST CONDITIONS

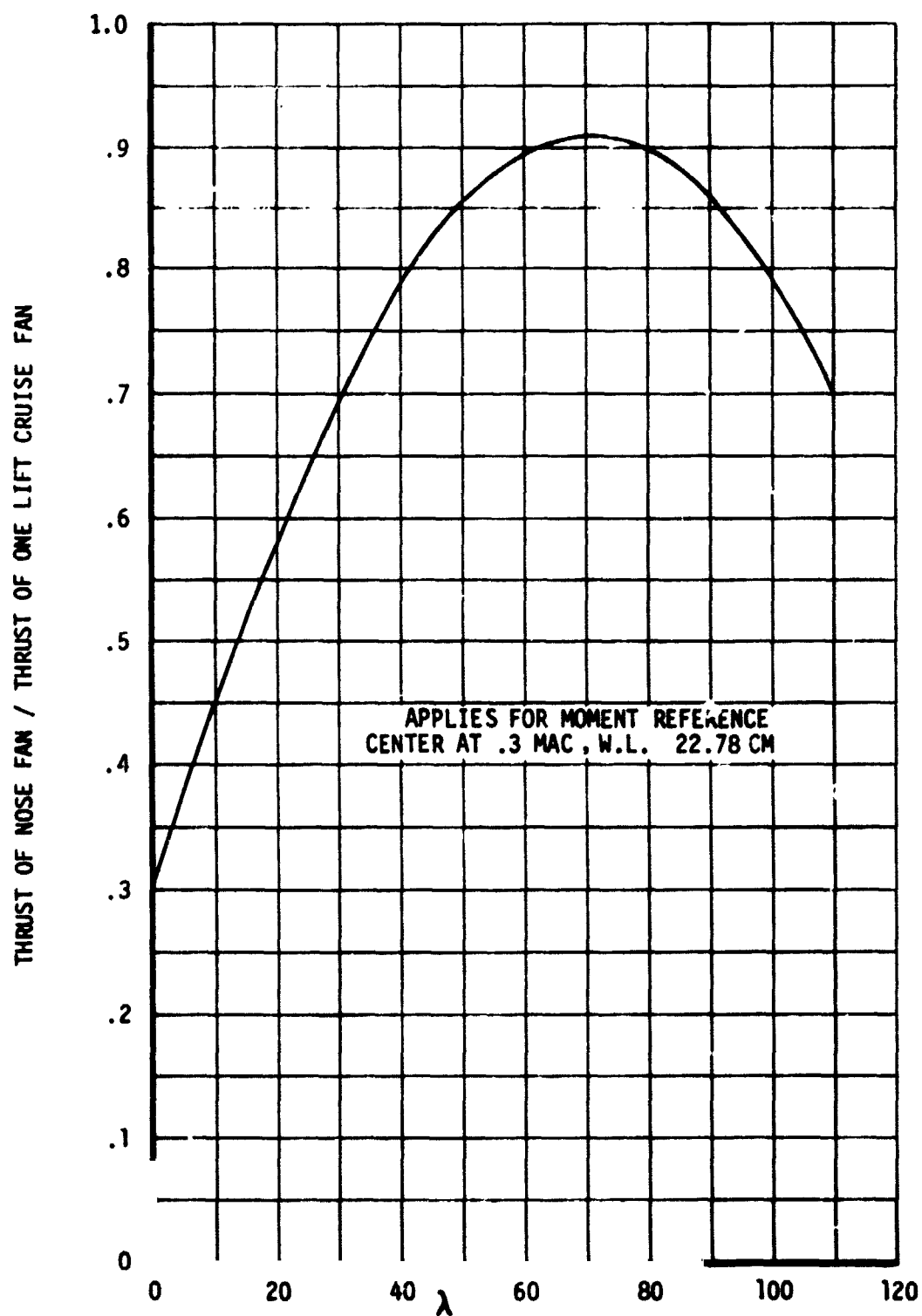


FIGURE 16 NOSE FAN THRUST REQUIRED FOR PITCHING MOMENT TRIM
IN STATIC FREE AIR HOVER

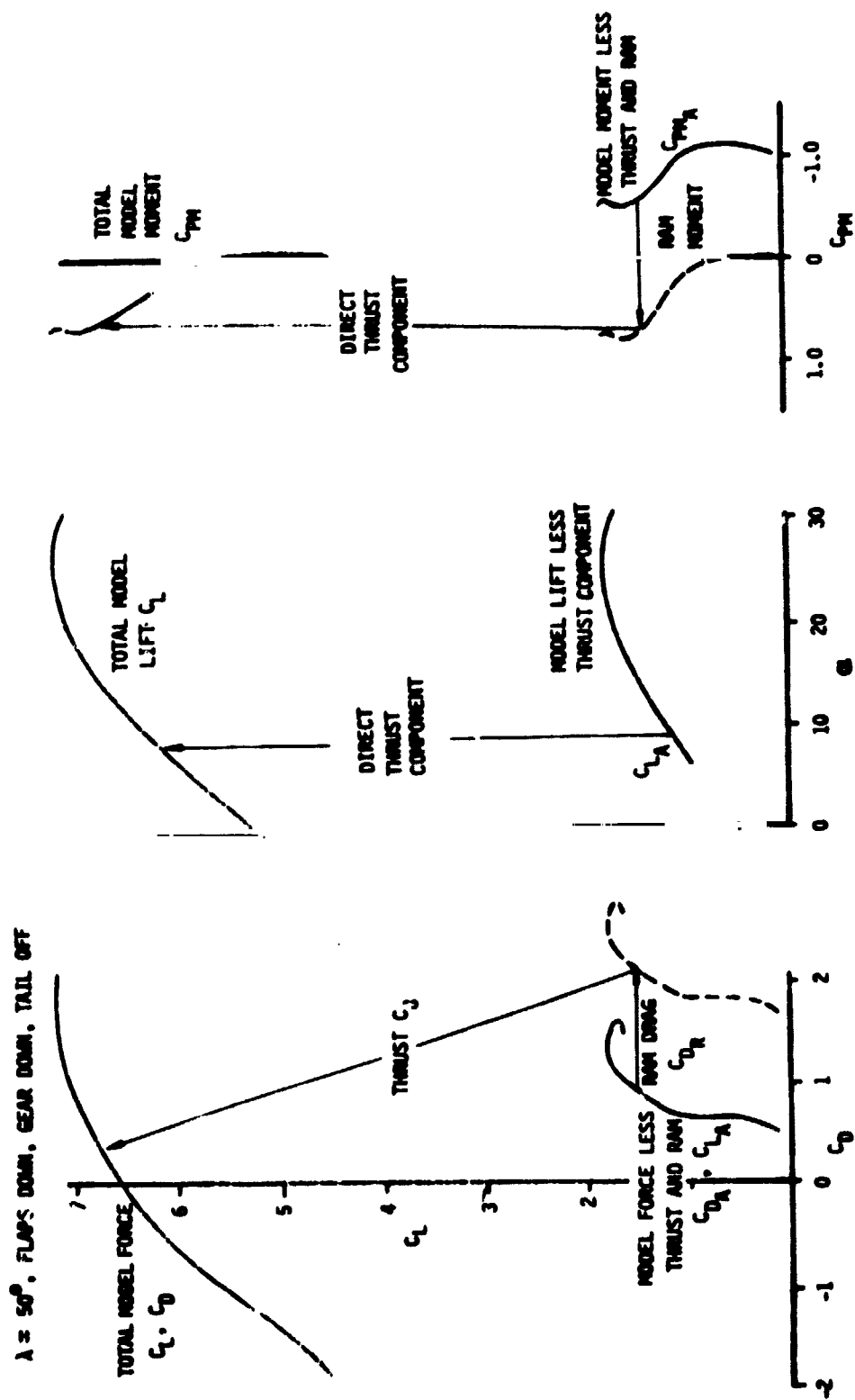


FIGURE 17 RELATIONSHIP OF AERODYNAMIC FORCES TO TOTAL FORCES

PUN	SYM	α
127	○	0°
132	□	0°
114	☆	0°
76	△	0°

$\lambda = 90^\circ$
 HOVER THRUST
 $V/V_j = 0$
 FLAPS DOWN
 GEAR DOWN
 $\delta = 0^\circ$

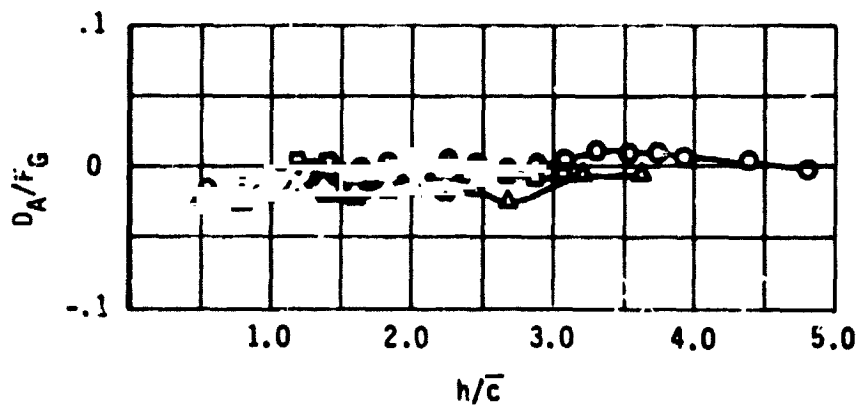
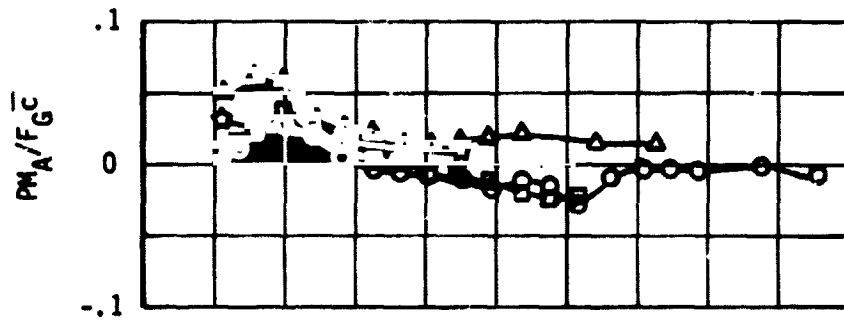
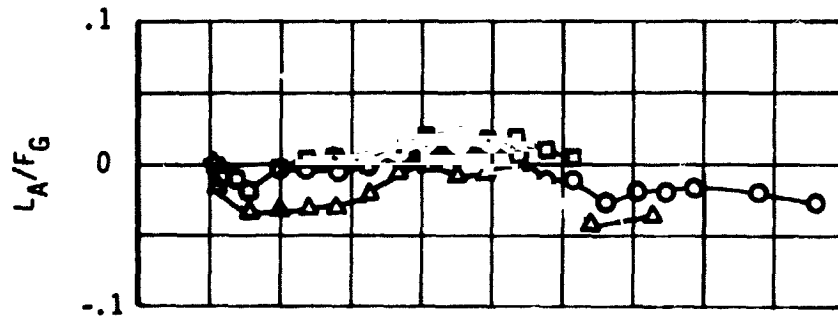
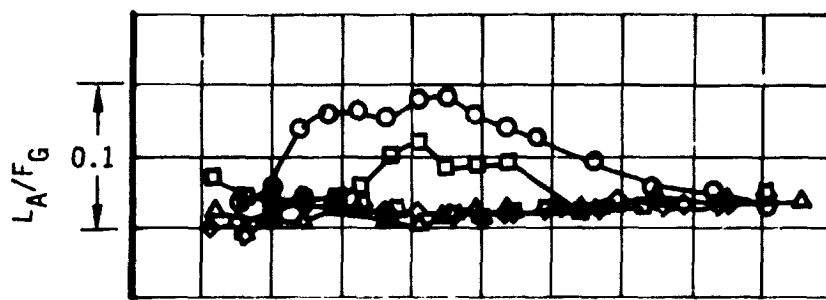


FIGURE 18 EFFECT OF GROUND HEIGHT ON INDUCED FORCES VTOL CONFIGURATION



RUN	SYM	α
79	○	-5°
76	□	0°
77	△	5°
80	◇	10°
81	☆	15°

$\lambda = 90^\circ$
 HOVER THRUST
 $V/V_j = 0$
 FLAPS DOWN
 GEAR DOWN
 $\alpha = 0^\circ$

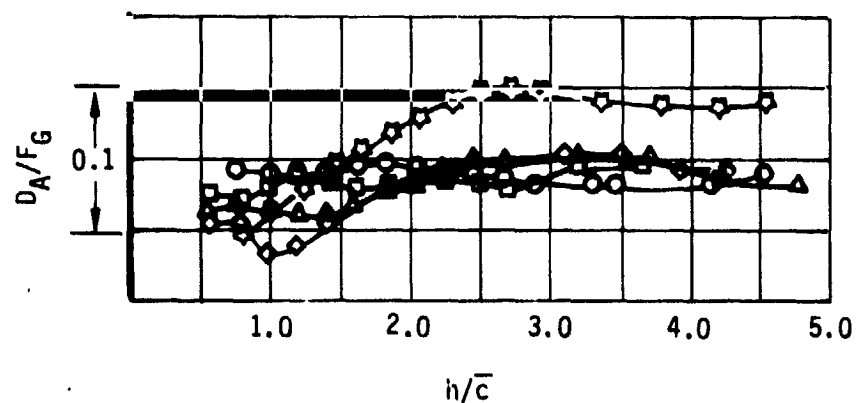
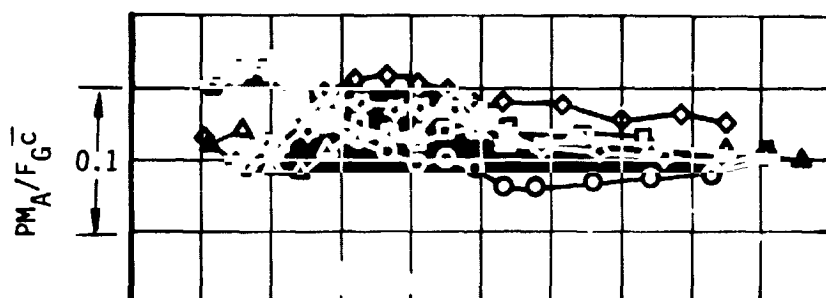
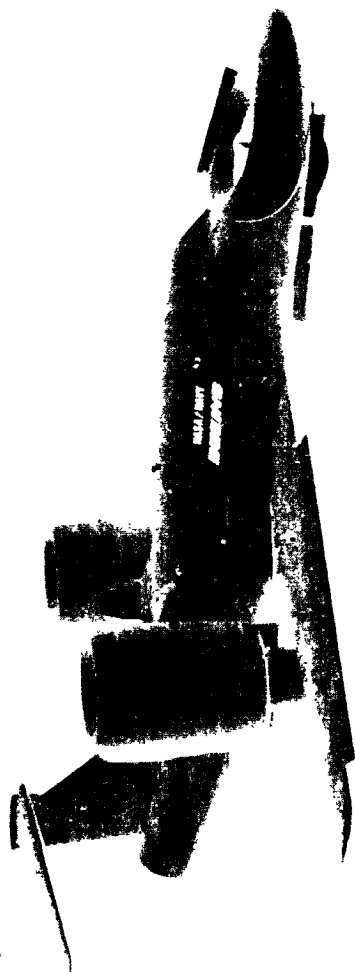


FIGURE 19 EFFECT OF ANGLE OF ATTACK ON INDUCED FORCES
VTOL CONFIGURATION, $\lambda = 90^\circ$

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TYPICAL OIL FLOW PATTERN ON GROUND PLANE
STATIC HOVER AT ALPHA = 5°

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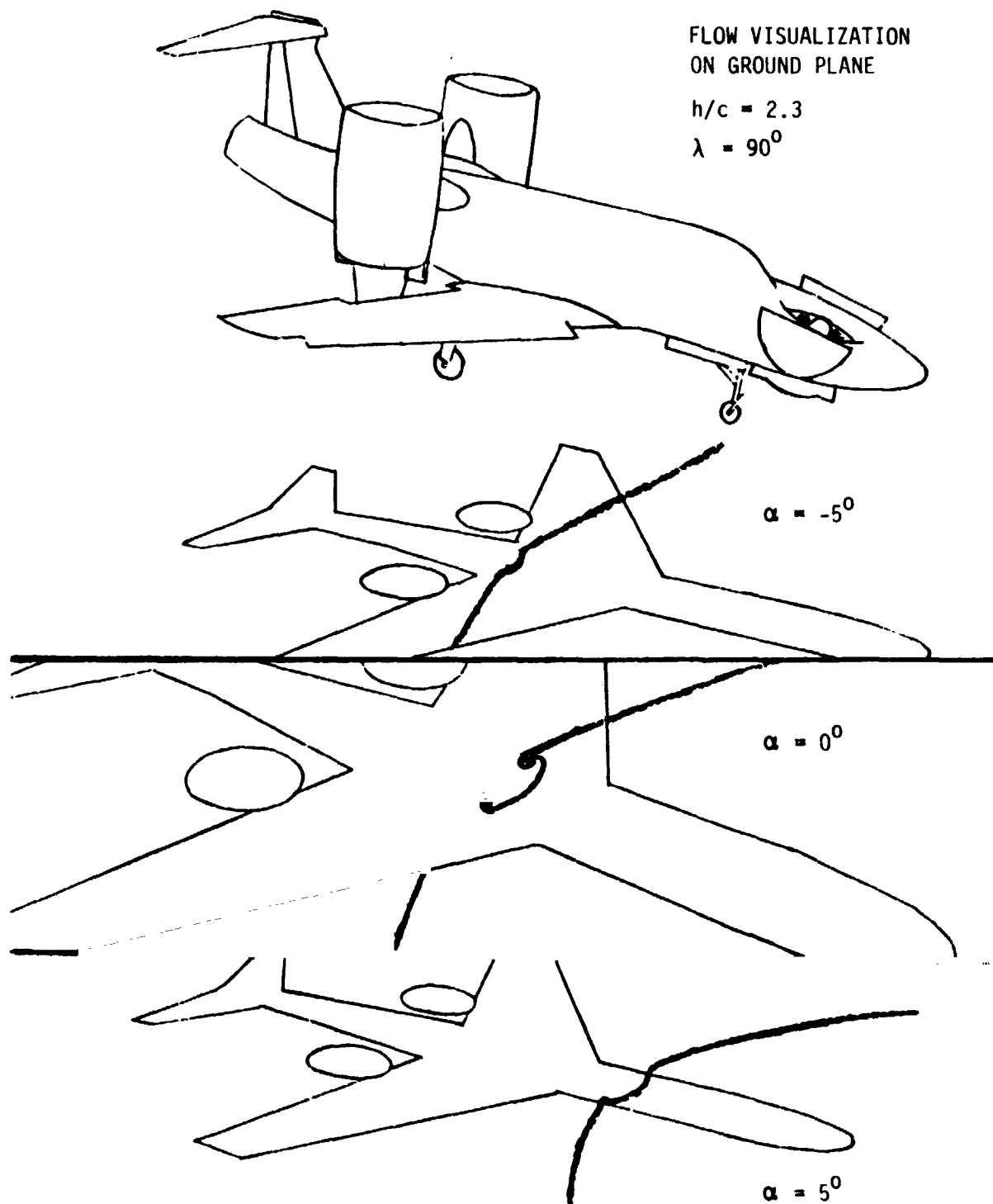


FIGURE 21 EFFECT OF ANGLE OF ATTACK ON
STATIC GROUND PLANE FLOW PATTERNS

RUN	SYM	α
125	○	0°
126	□	10°

$\lambda = 80^\circ$
 HOVER THRUST
 $V/V_j = 0$
 FLAPS DOWN
 GEAR DOWN
 $\delta = 0^\circ$

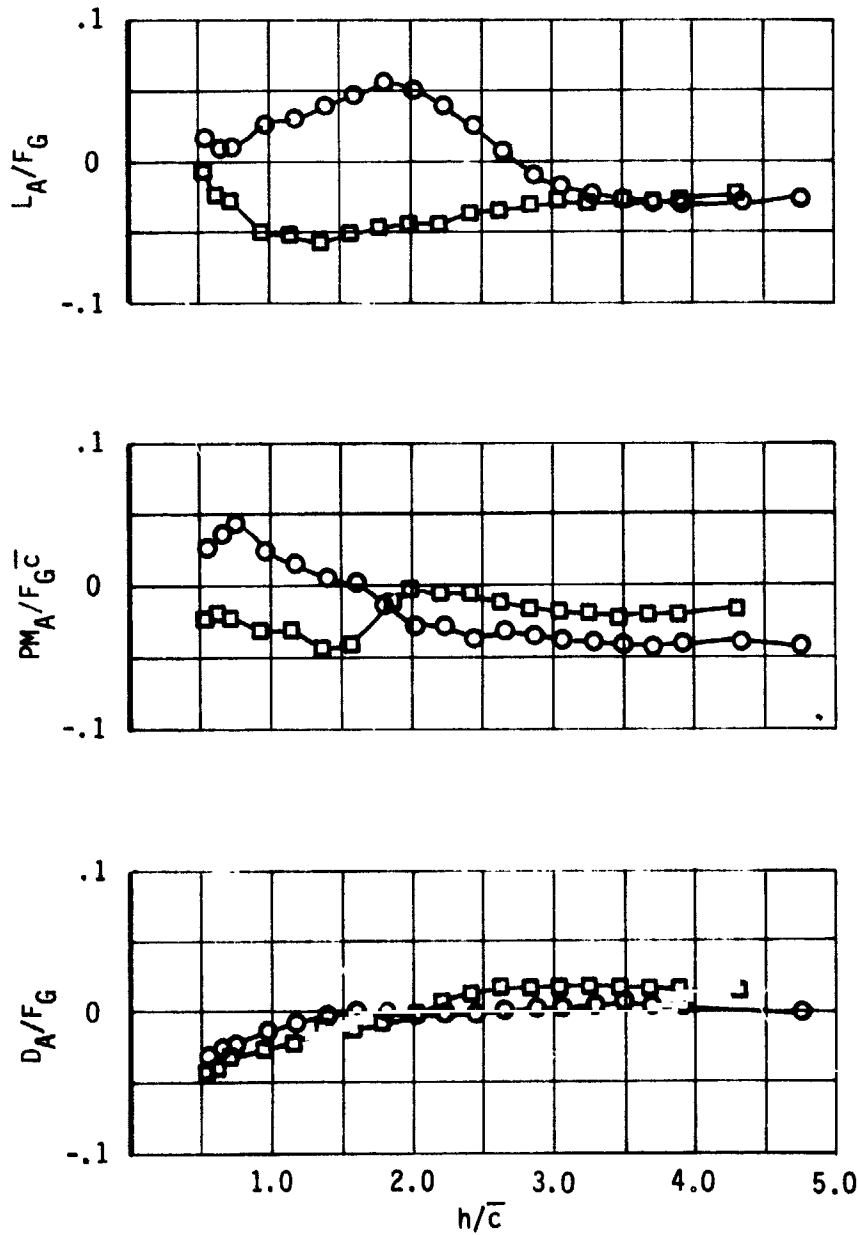


FIGURE 22 EFFECT OF ANGLE OF ATTACK
VTOL CONFIGURATION, $\lambda = 80^\circ$

RUN	SYM	λ
125	○	80°
127	□	90°
123	△	95

$\alpha = 0^\circ$
 HOVER THRUST
 $V/V_j = 0$
 FLAPS DOWN
 GEAR DOWN
 $\delta = 0^\circ$

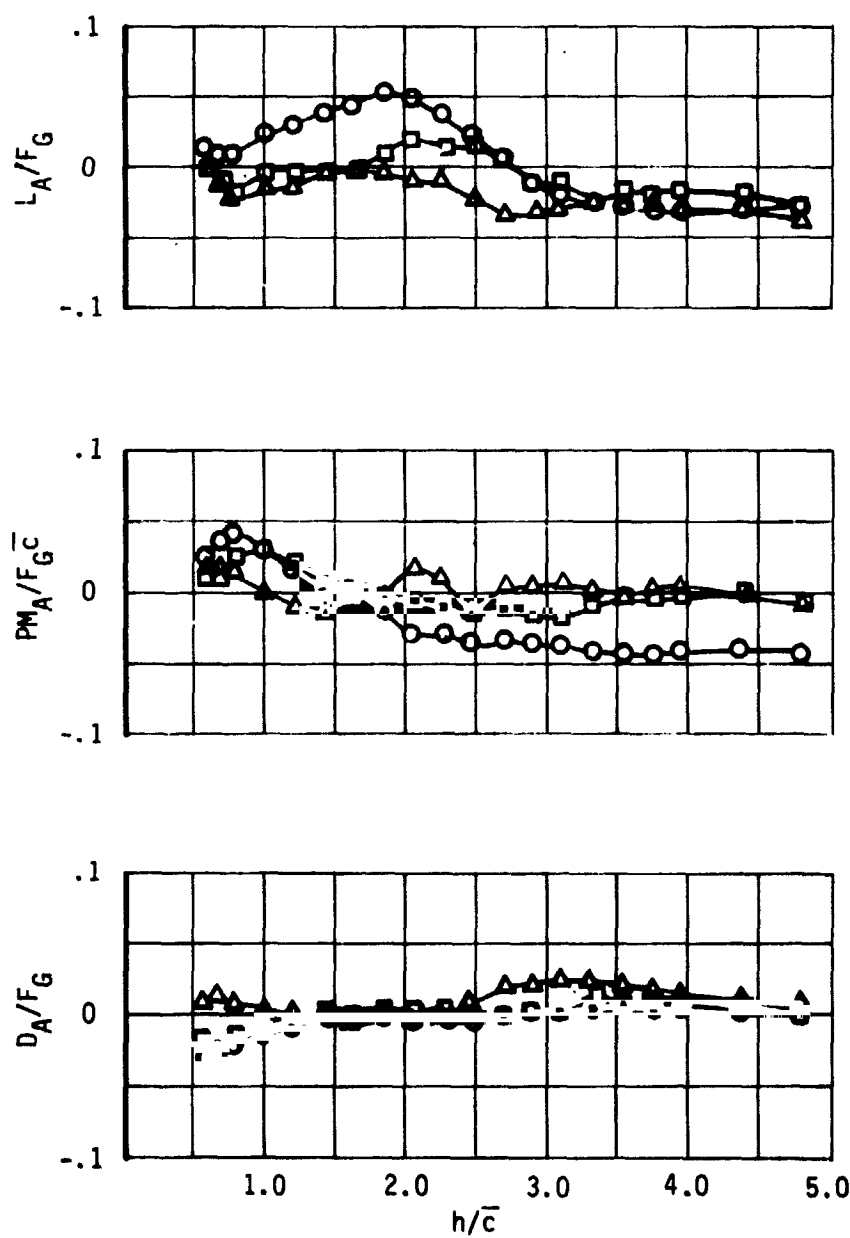


FIGURE 23 EFFECT OF NACELLE TILT ANGLE VTOL CONFIGURATION

RUN	SYM	ϕ
76	○	0°
84	△	-5°
83	□	-10°
85	◇	-15°

$\alpha = 0^\circ$
 $\lambda = 90^\circ$
 HOVER THRUST
 $V/V_j = 0$
 FLAPS DOWN
 GEAR DOWN
 $\delta = 0^\circ$

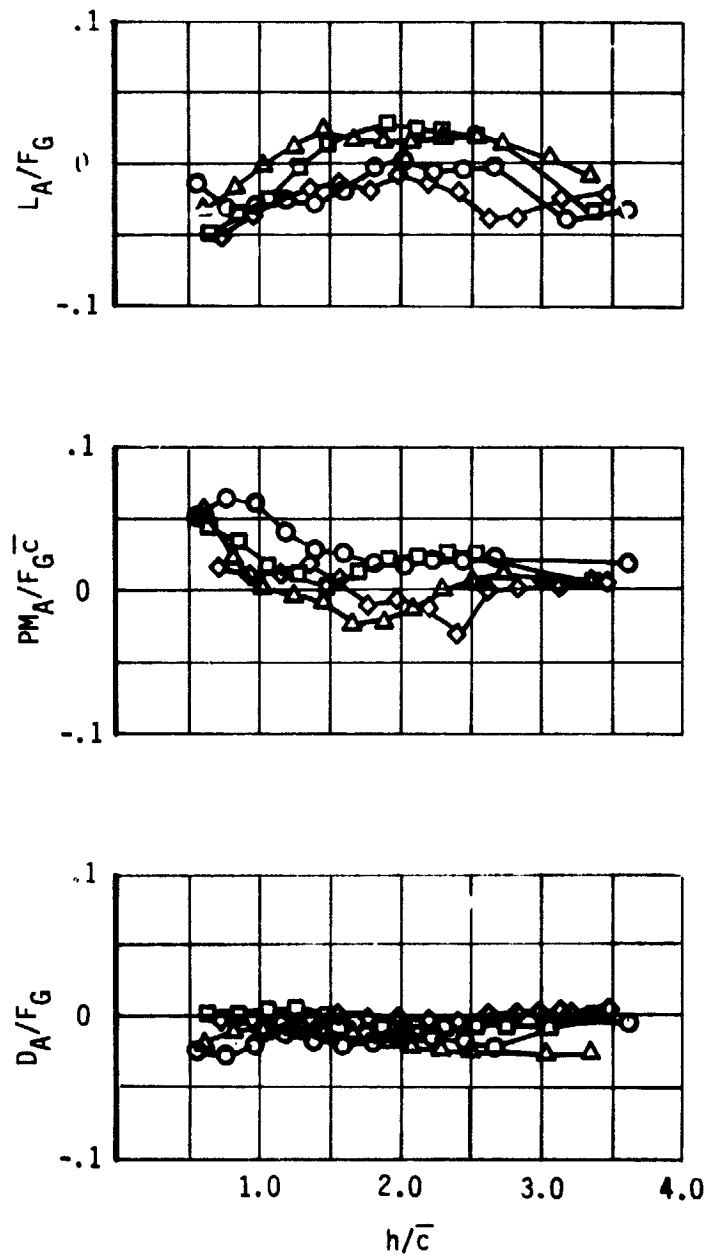


FIGURE 24a EFFECT OF ROLL ANGLE
VTOL CONFIGURATION

RUN	SYM	δ
76	○	0°
84	△	-5°
83	□	-10°
85	◇	-15°

$\alpha = 0^\circ$
 $\lambda = 90$
 HOVER THRUST
 $V/V_j = 0$
 FLAPS DOWN
 GEAR DOWN
 $\delta = 0^\circ$

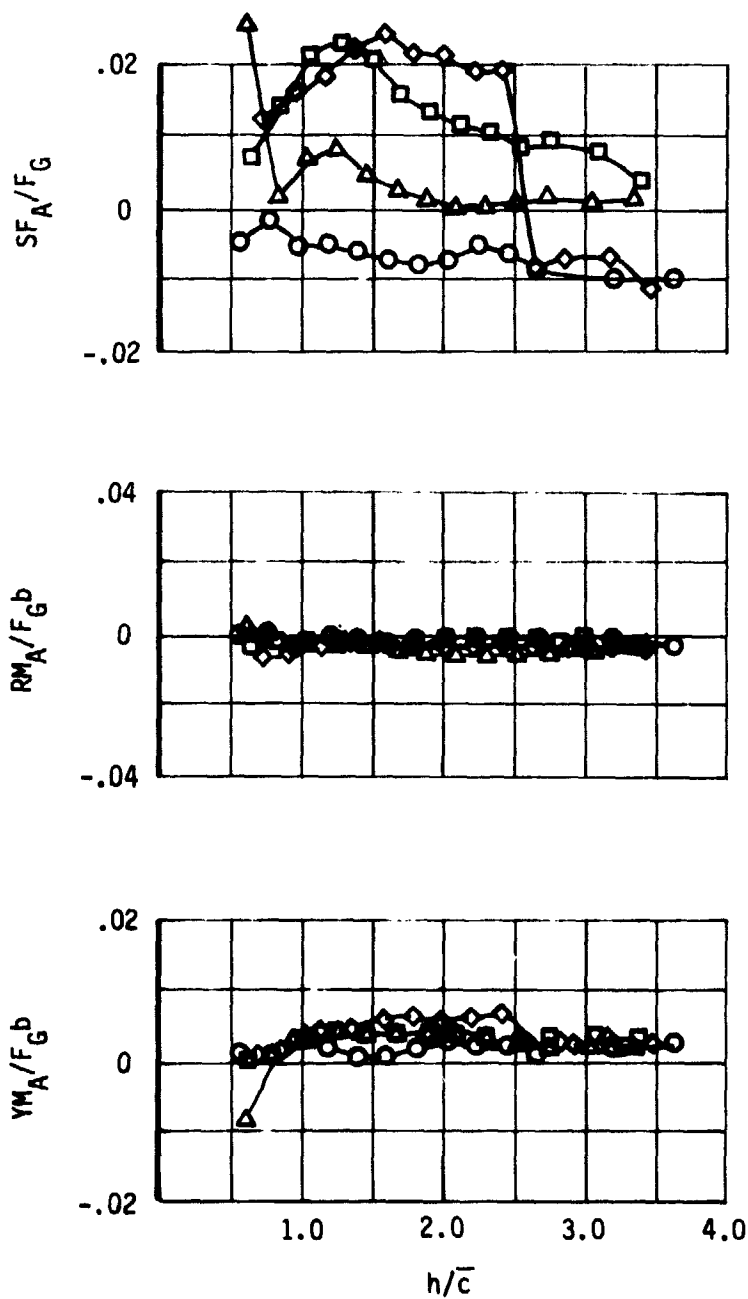


FIGURE 24b EFFECT OF ROLL ANGLE
 VTOL CONFIGURATION

RUN	SYM	α
83	○	0°
88	□	5°
90	◇	10°

◆ = -10°
 $\lambda = 90^\circ$
 HOVER THRUST
 $V/V_j = 0$
 FLAPS DOWN
 GEAR DOWN
 ▲ = 0°

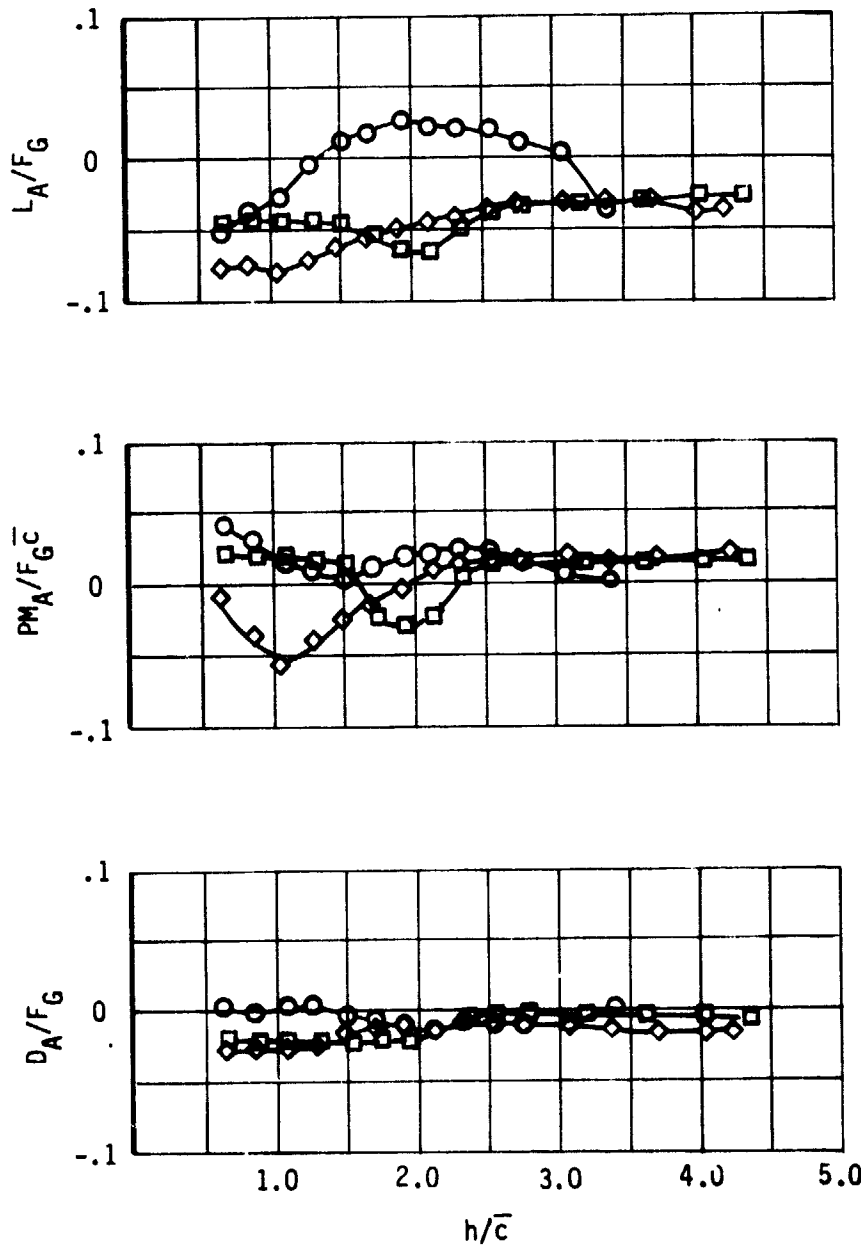


FIGURE 25a EFFECT OF ANGLE OF ATTACK WITH ROLL
VTOL CONFIGURATION"

RUN	SYM	α
83	○	0°
88	□	5°
90	◇	10°

$\phi = -10^\circ$
 $\lambda = 90^\circ$
 HOVER THRUST
 $V/V_j = 0$
 FLAPS DOWN
 GEAR DOWN
 $\delta = 0^\circ$

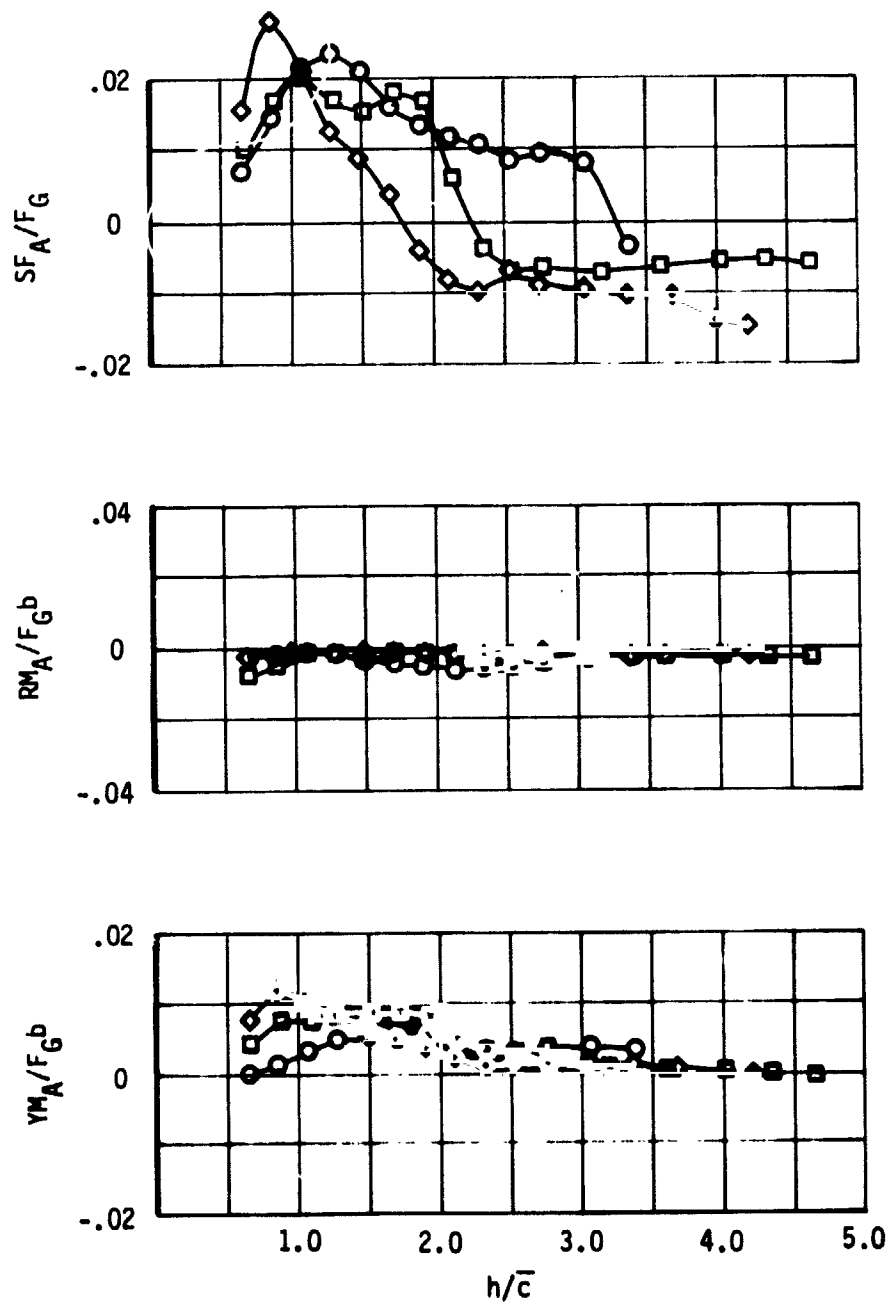


FIGURE 25b EFFECT OF ANGLE OF ATTACK WITH ROLL
VTOL CONFIGURATION

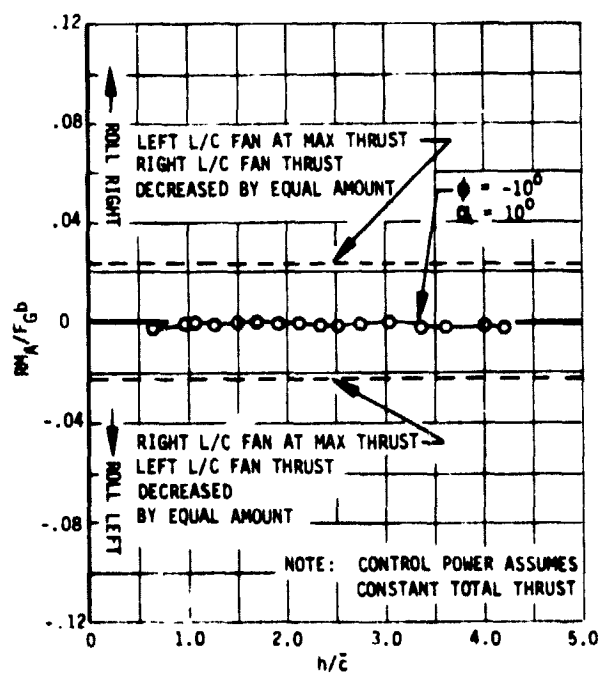
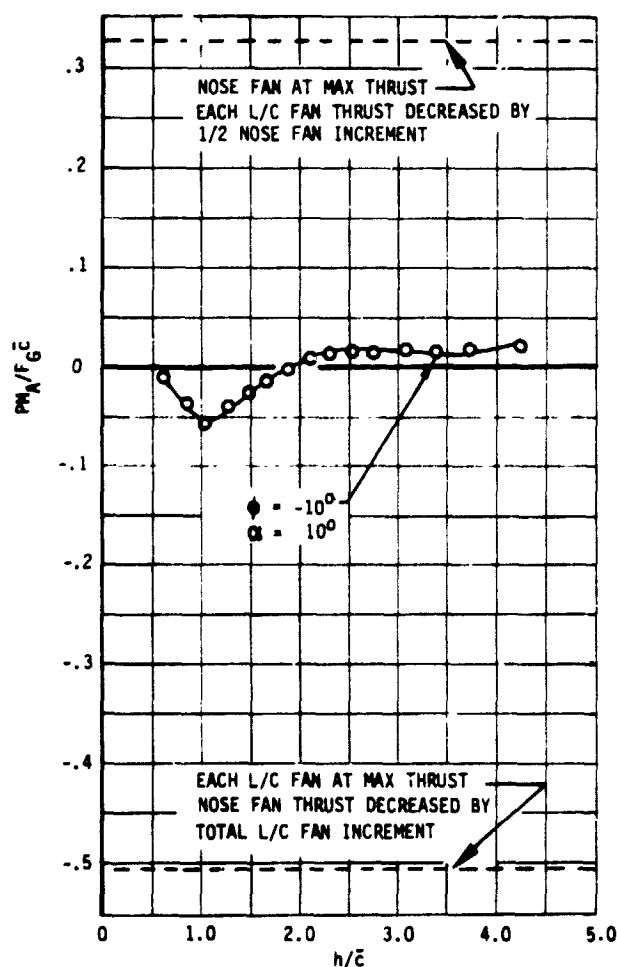


FIGURE 26 COMPARISON OF TYPICAL INDUCED PITCH AND ROLL MOMENTS
IN STATIC HOVER WITH AVAILABLE PITCH AND ROLL CONTROL POWER

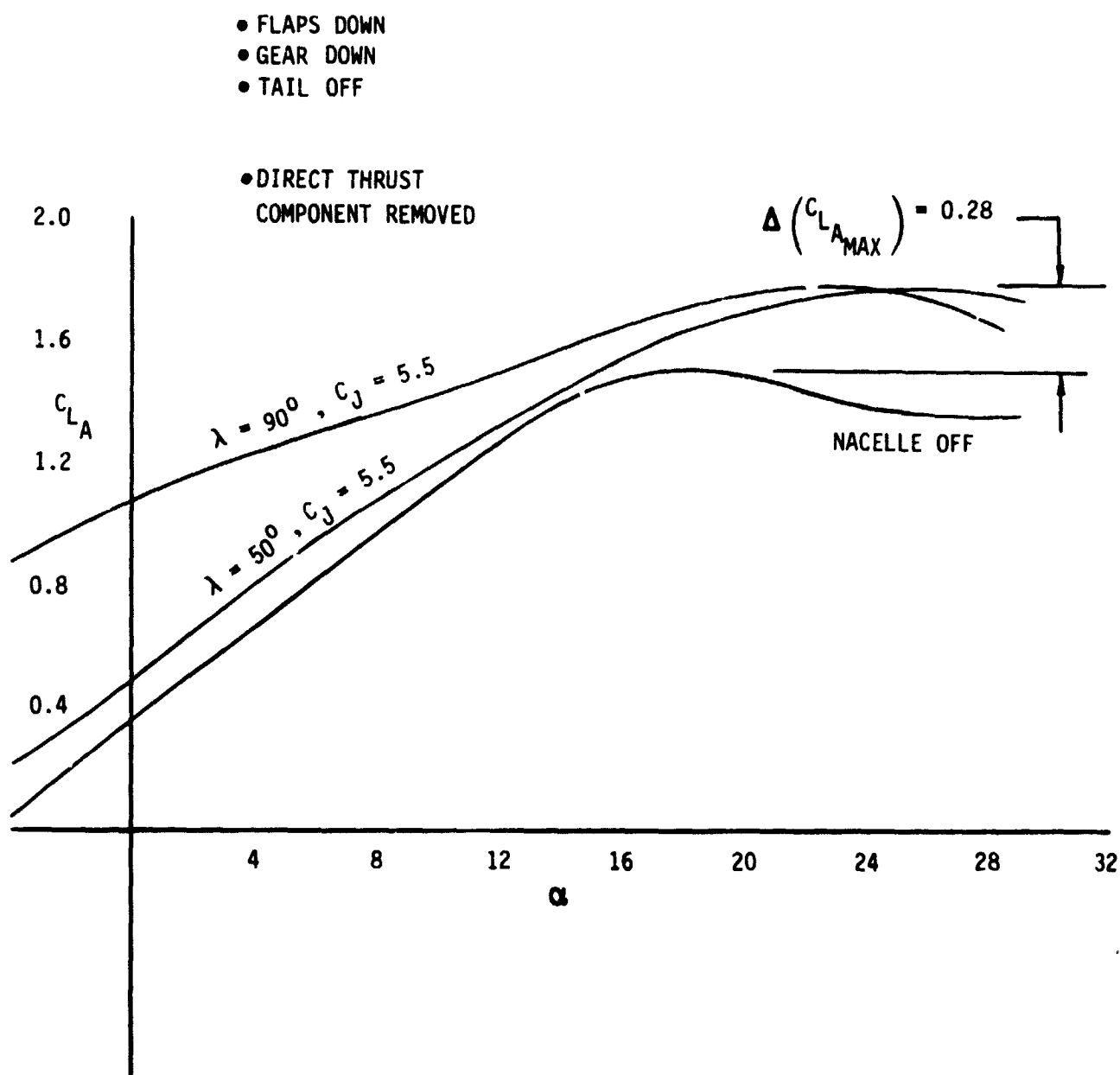


FIGURE 27 INDUCED EFFECT OF PROPULSION SYSTEM
ON LIFT AT $C_J = 5.5$

- FLAPS DOWN
- GEAR DOWN
- TAIL OFF
- DIRECT THRUST COMPONENT REMOVED
- $\alpha = 8^\circ$

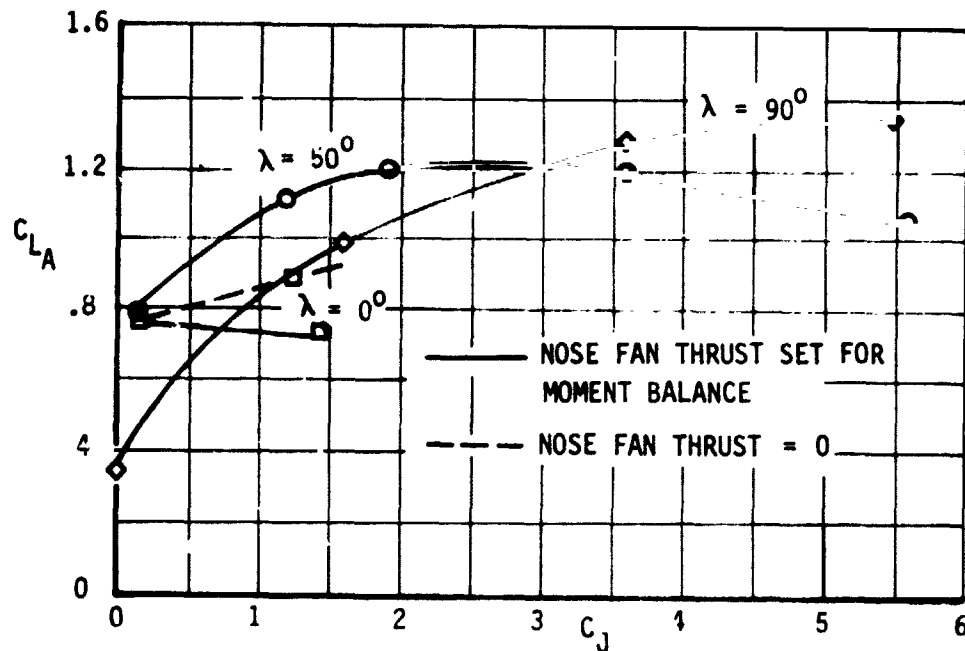


FIGURE 28 EFFECT OF THRUST COEFFICIENT ON LIFT

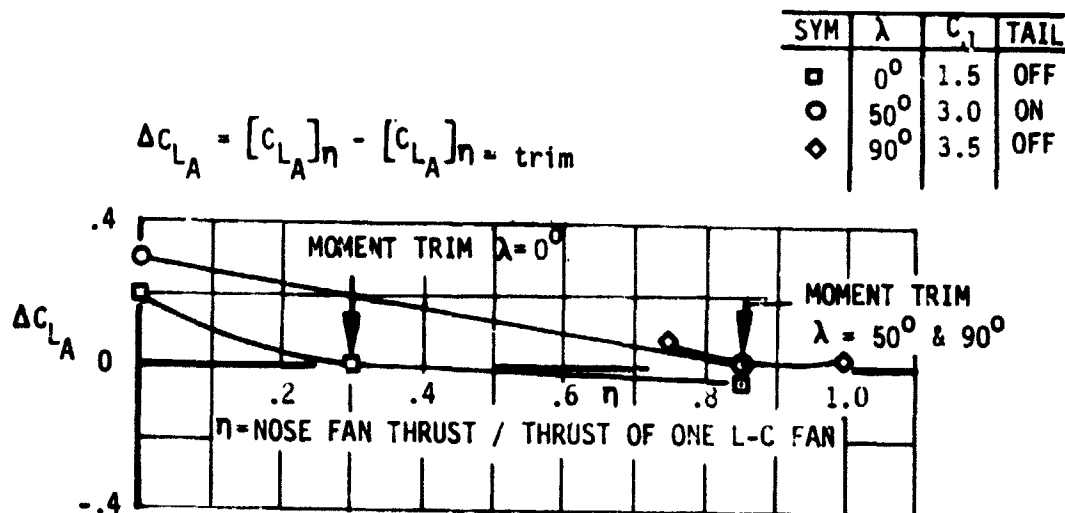
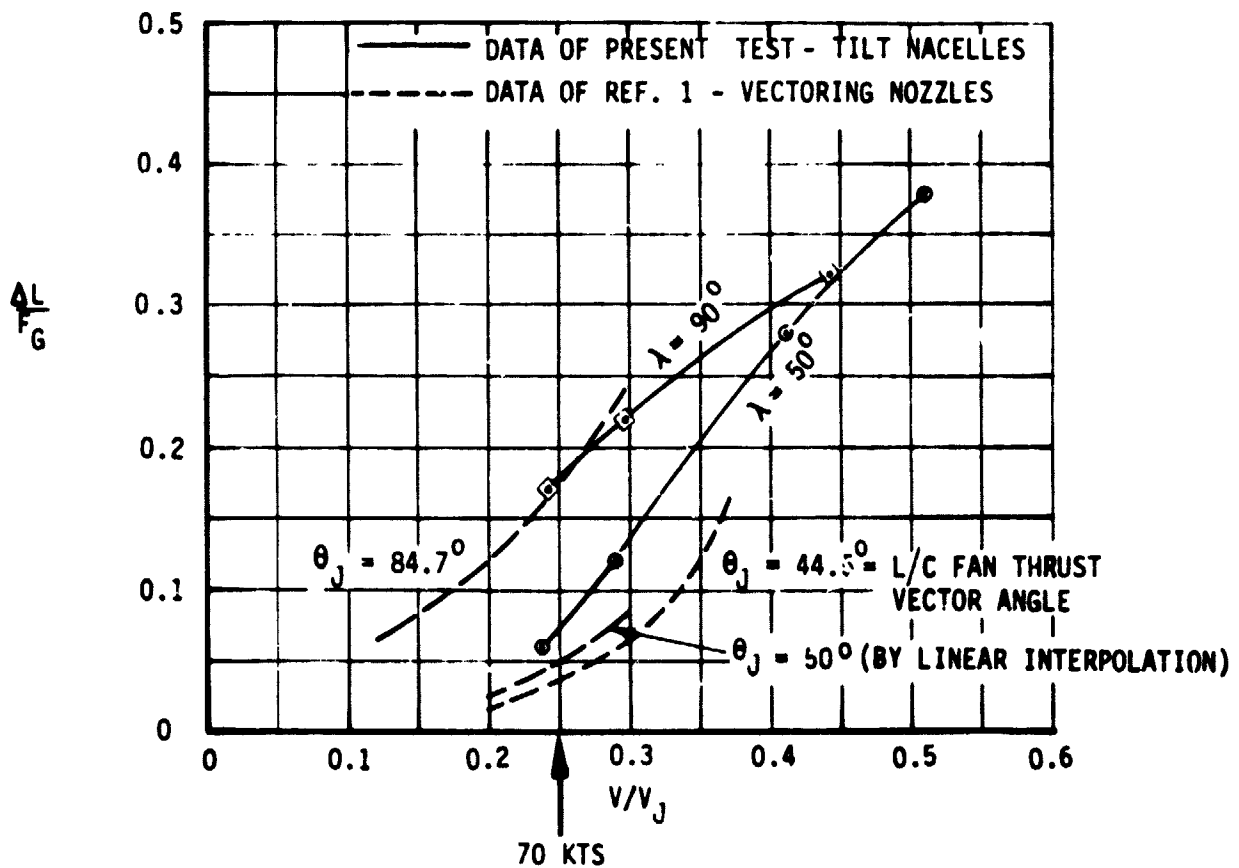


FIGURE 29 EFFECT OF THRUST BALANCE ON LIFT

- $\alpha = 0^\circ$
- FLAPS DOWN
- GEAR DOWN
- TAIL OFF



- NOTES
- (1) $\Delta L = (L_A) - (L_A)_{\text{REFERENCE}}$
 - (2) L_A = LIFT WITH DIRECT THRUST REMOVED
 - (3) $(L_A)_{\text{REFERENCE}}$ IS BASED ON:
 $\lambda = 0^\circ$, C_J = RAM, NOSE FAN COVERED,
 FLAPS DOWN, GEAR DOWN, TAIL OFF

FIGURE 30 A SUMMARY OF INDUCED EFFECT OF PROPULSION SYSTEM ON LIFT

- FLAPS DOWN
- GEAR DOWN
- TAIL OFF
- DIRECT THRUST COMPONENT REMOVED

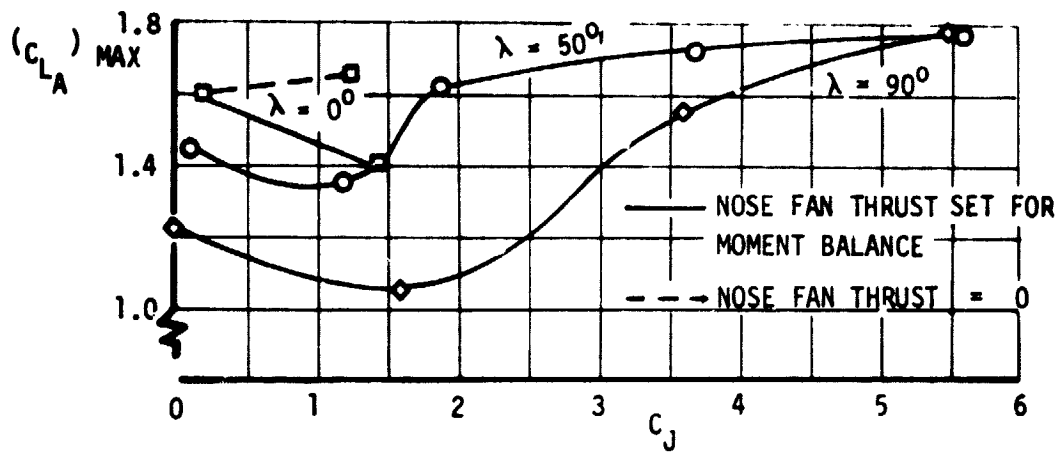


FIGURE 31 EFFECT OF THRUST COEFFICIENT ON MAXIMUM LIFT

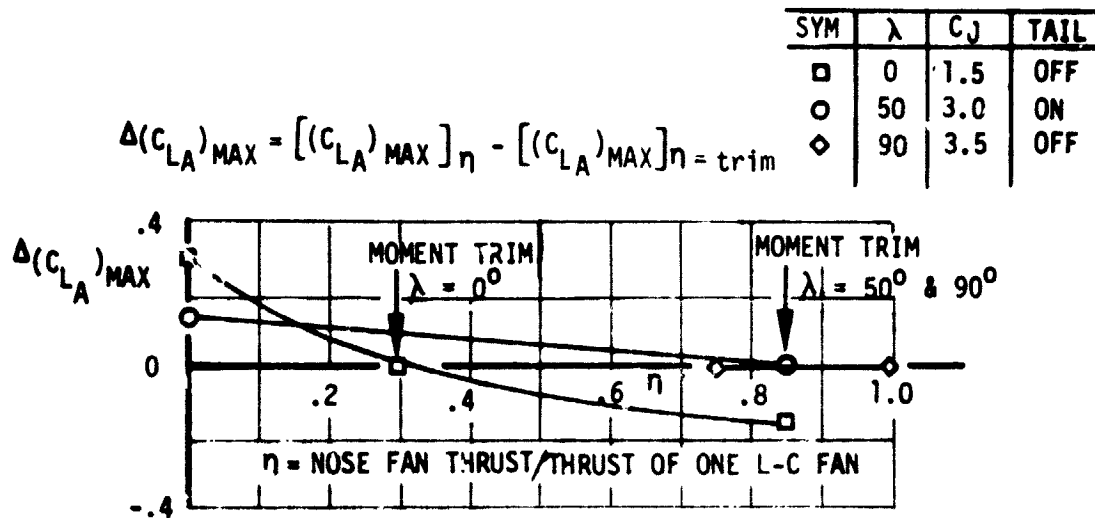


FIGURE 32 EFFECT OF THRUST BALANCE ON MAXIMUM LIFT

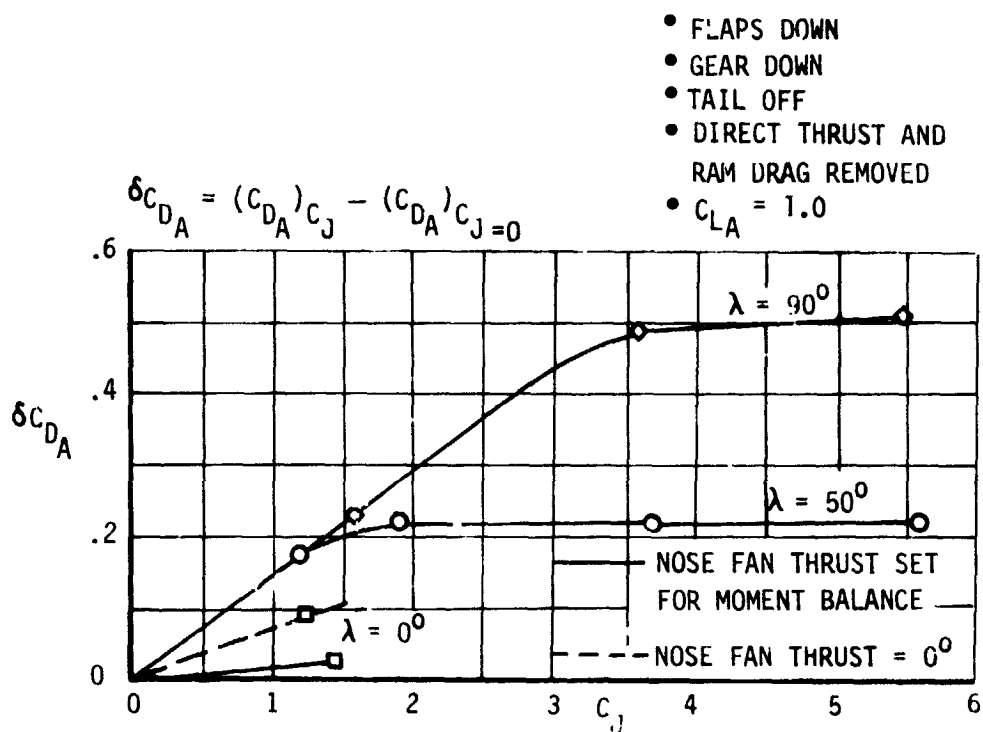


FIGURE 33 EFFECT OF THRUST COEFFICIENT ON DRAG

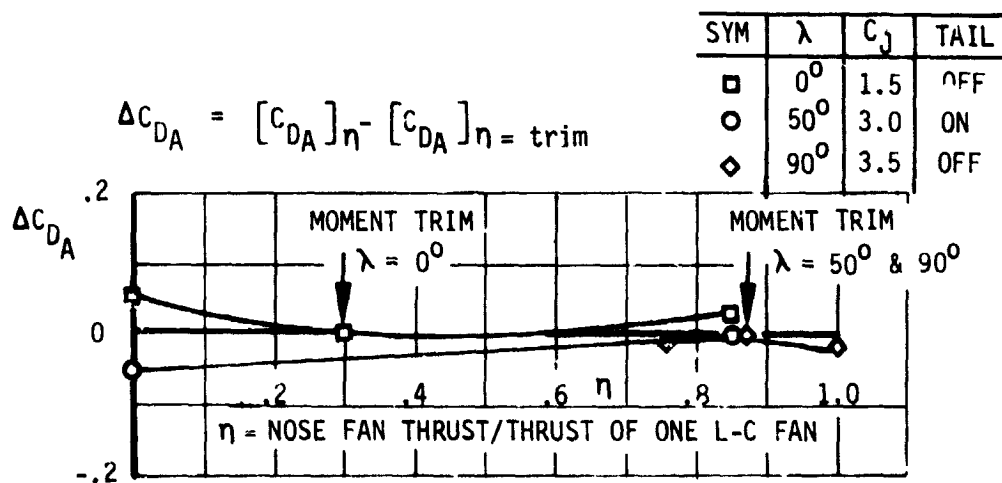


FIGURE 34 EFFECT OF THRUST BALANCE ON DRAG

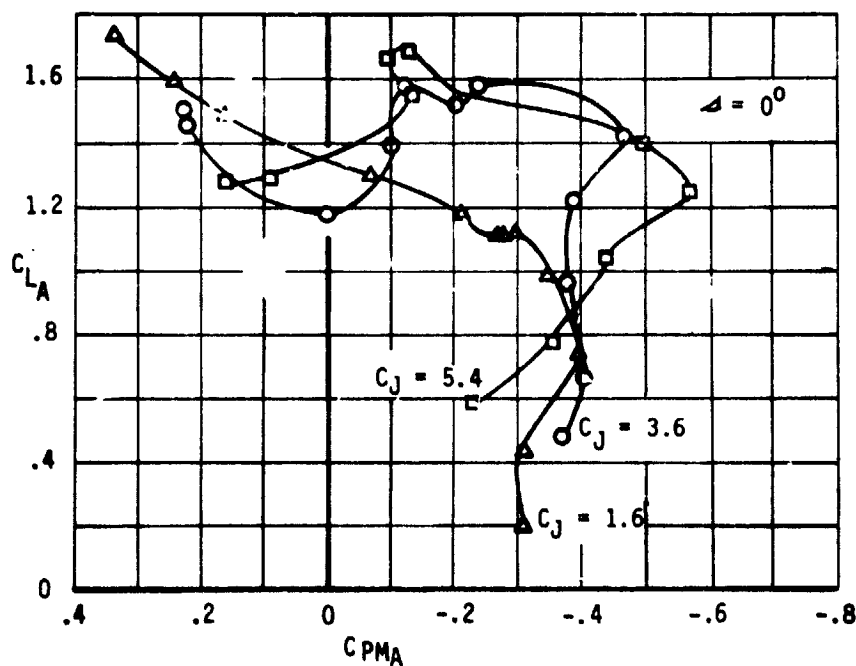
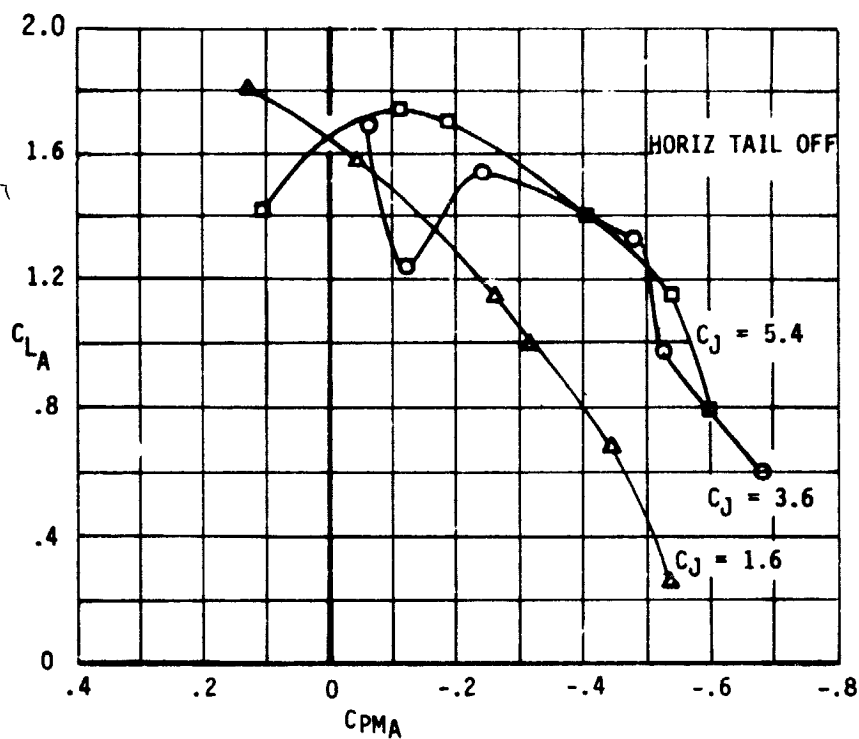
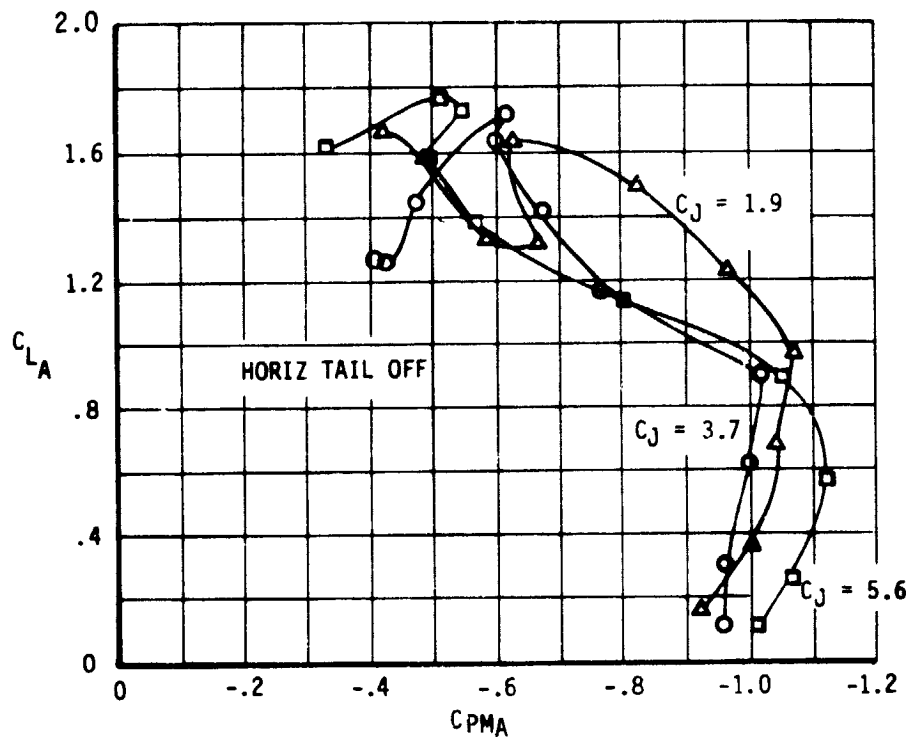


FIGURE 35 EFFECT OF THRUST COEFFICIENT AND HORIZONTAL TAIL ON LONGITUDINAL STABILITY - $\gamma = 90^\circ$



- FLAPS DOWN
- GEAR DOWN
- DIRECT THRUST AND RAM COMPONENTS REMOVED

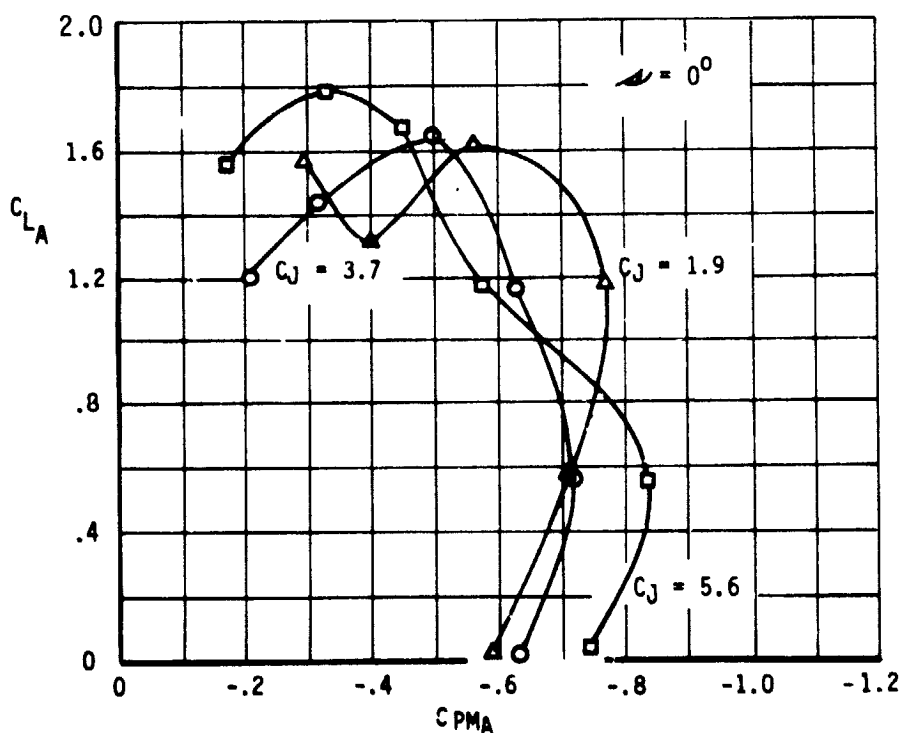


FIGURE 36 EFFECT OF THRUST COEFFICIENT AND HORIZONTAL TAIL ON LONGITUDINAL STABILITY - $\lambda = 50^\circ$

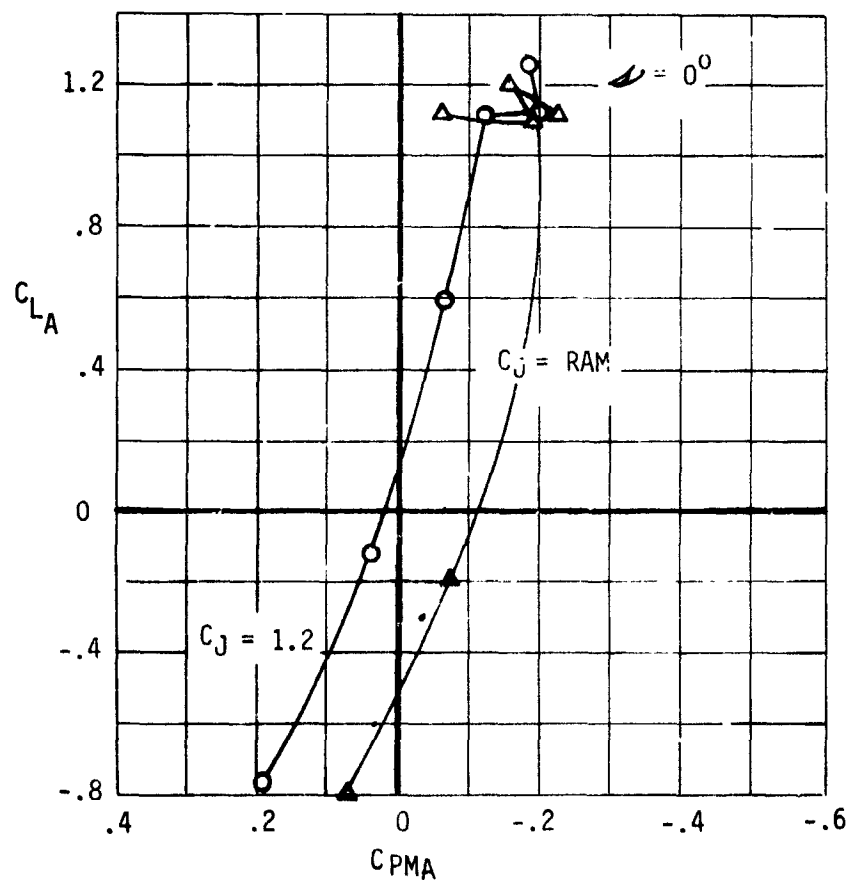
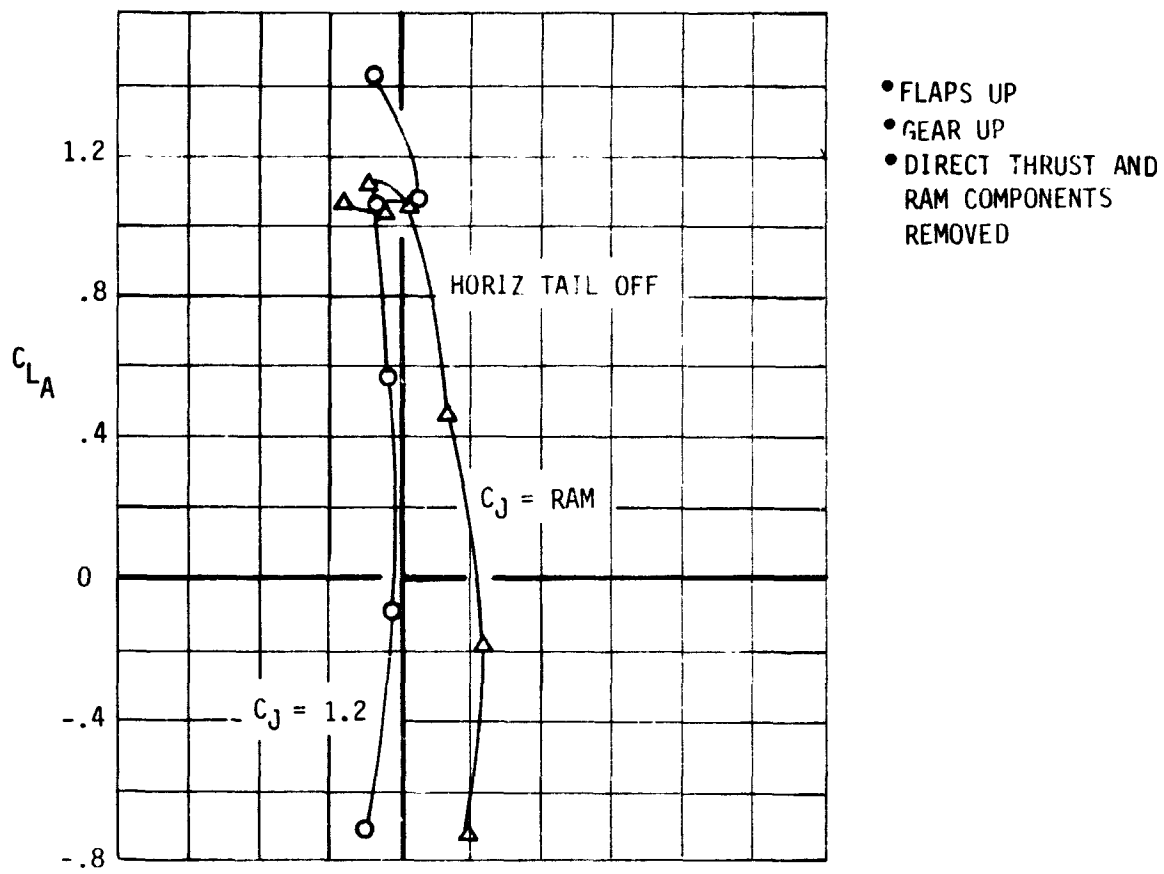
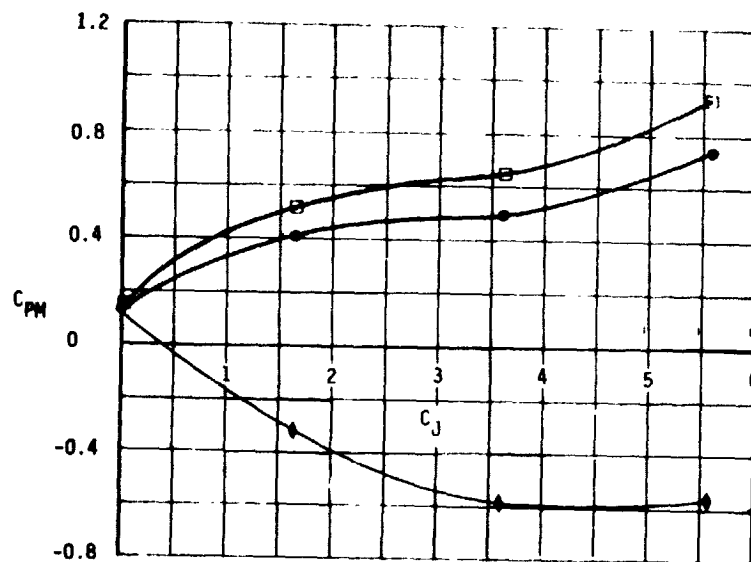


FIGURE 37 EFFECT OF THRUST COEFFICIENT AND HORIZONTAL TAIL ON LONGITUDINAL STABILITY - $\lambda = 0^\circ$

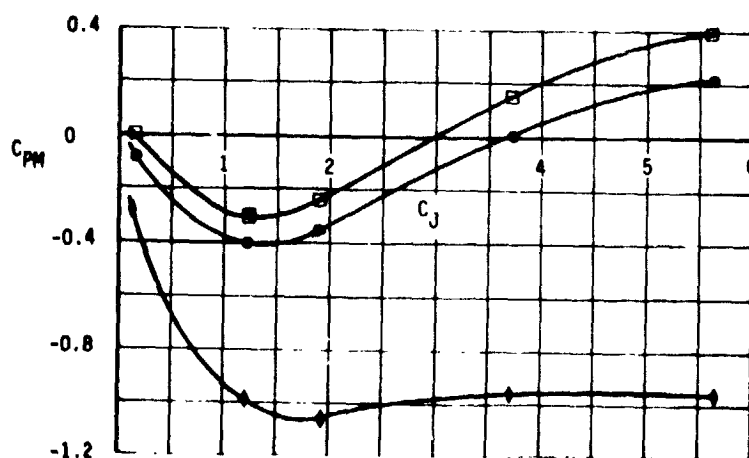
- FLAPS DOWN
- GEAR DOWN
- TAIL OFF
- $C_{L_A} = 1.0$

- C_{PM} (TOTAL MEASURED MODEL MOMENTS)
- C_{PM} (TOTAL MOMENTS WITH RAM MOMENTS CORRECTED) TO FULL SCALE
- ♦ $C_{PM_A} = C_{PM}$ LESS CALCULATED RAM MOMENT AND DIRECT THRUST MOMENTS (MODEL DATA)



$\lambda = 90^\circ$

NOSE FAN THRUST SET
FOR STATIC MOMENT TRIM
(SEE FIGURE 16)

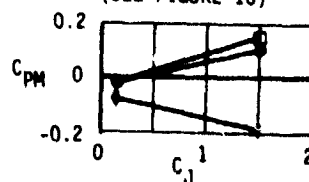
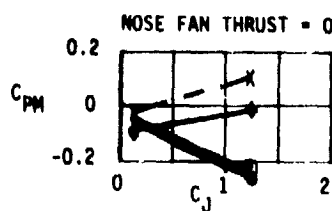


$\lambda = 50^\circ$

NOSE FAN THRUST SET
FOR STATIC MOMENT TRIM
(SEE FIGURE 16)

--X-- C_{PM_A} PLUS MODEL RAM MOMENTS

NOSE FAN THRUST SET
FOR STATIC MOMENT TRIM
(SEE FIGURE 16)



$\lambda = 0^\circ$

FIGURE 38 EFFECT OF THRUST COEFFICIENT ON PITCHING MOMENT

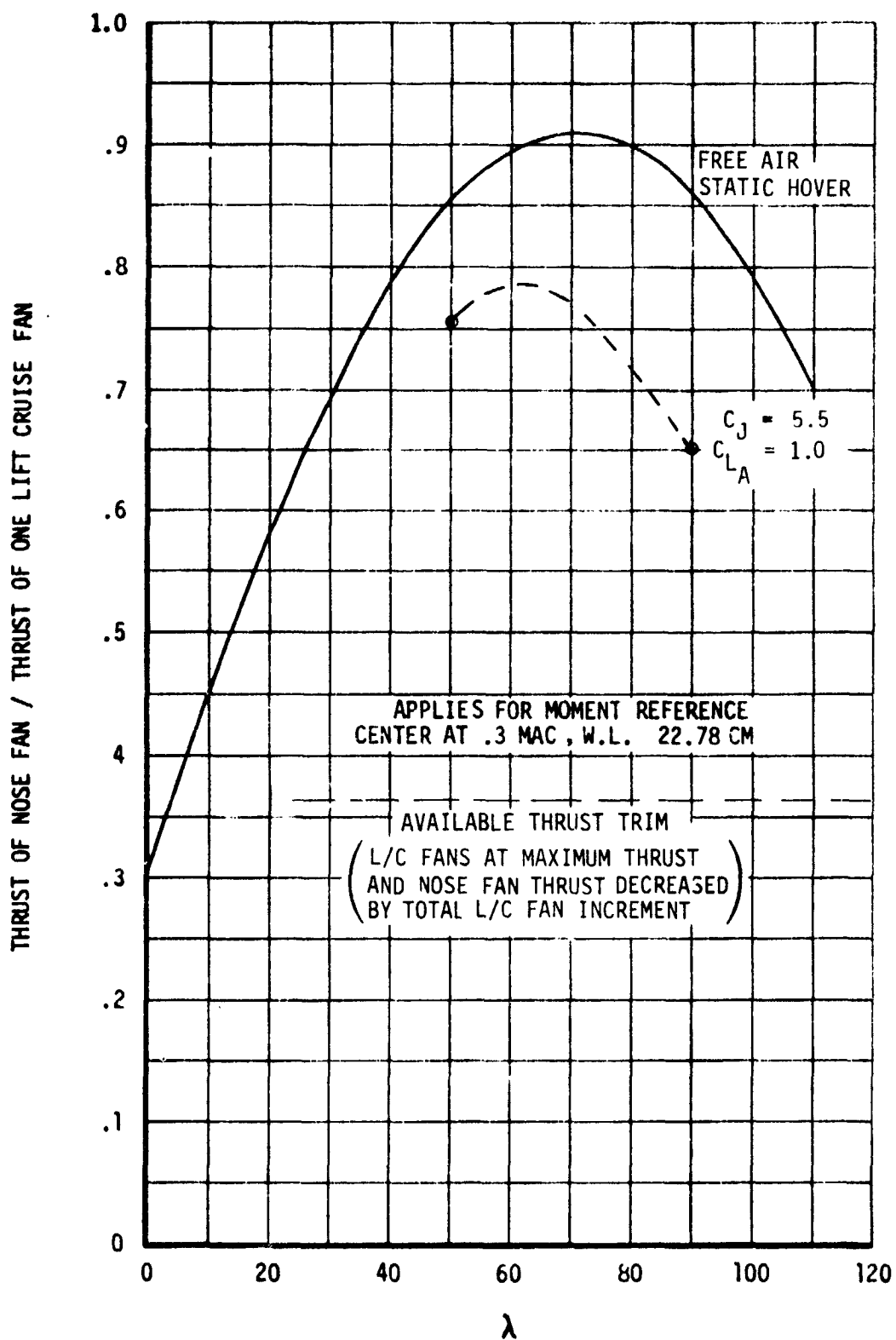


FIGURE 39 NOSE FAN THRUST REQUIRED FOR PITCHING MOMENT
TRIM, TAIL OFF, FLAPS DOWN

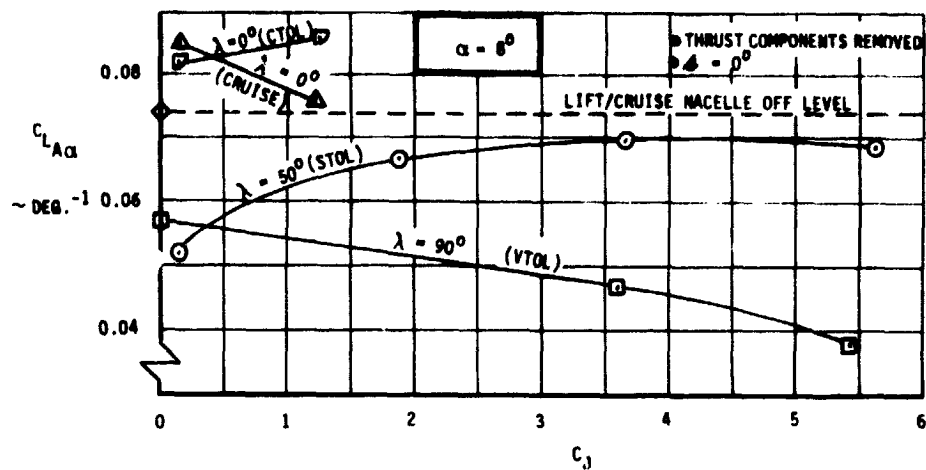


FIGURE 40 EFFECT OF THRUST ON LIFT CURVE SLOPE

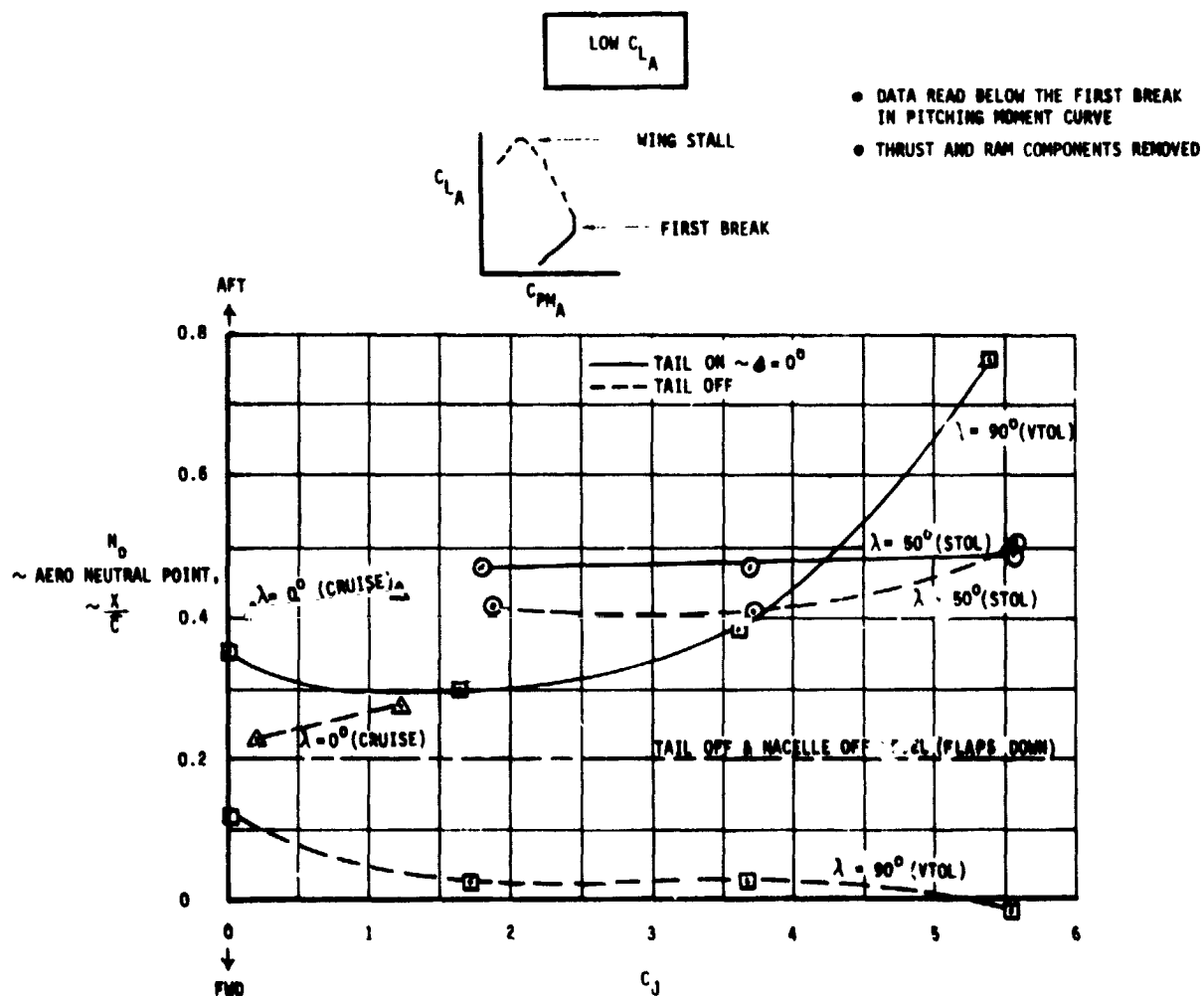


FIGURE 41 EFFECT OF THRUST ON NEUTRAL POINT LOCATION (LOW C_{L_A})

HIGH C_{LA}

- DATA READ ABOVE FIRST BREAK IN PITCHING MOMENT CURVE BUT BELOW THE BREAK FOR WING STALL
- THRUST AND RAM FORCES REMOVED

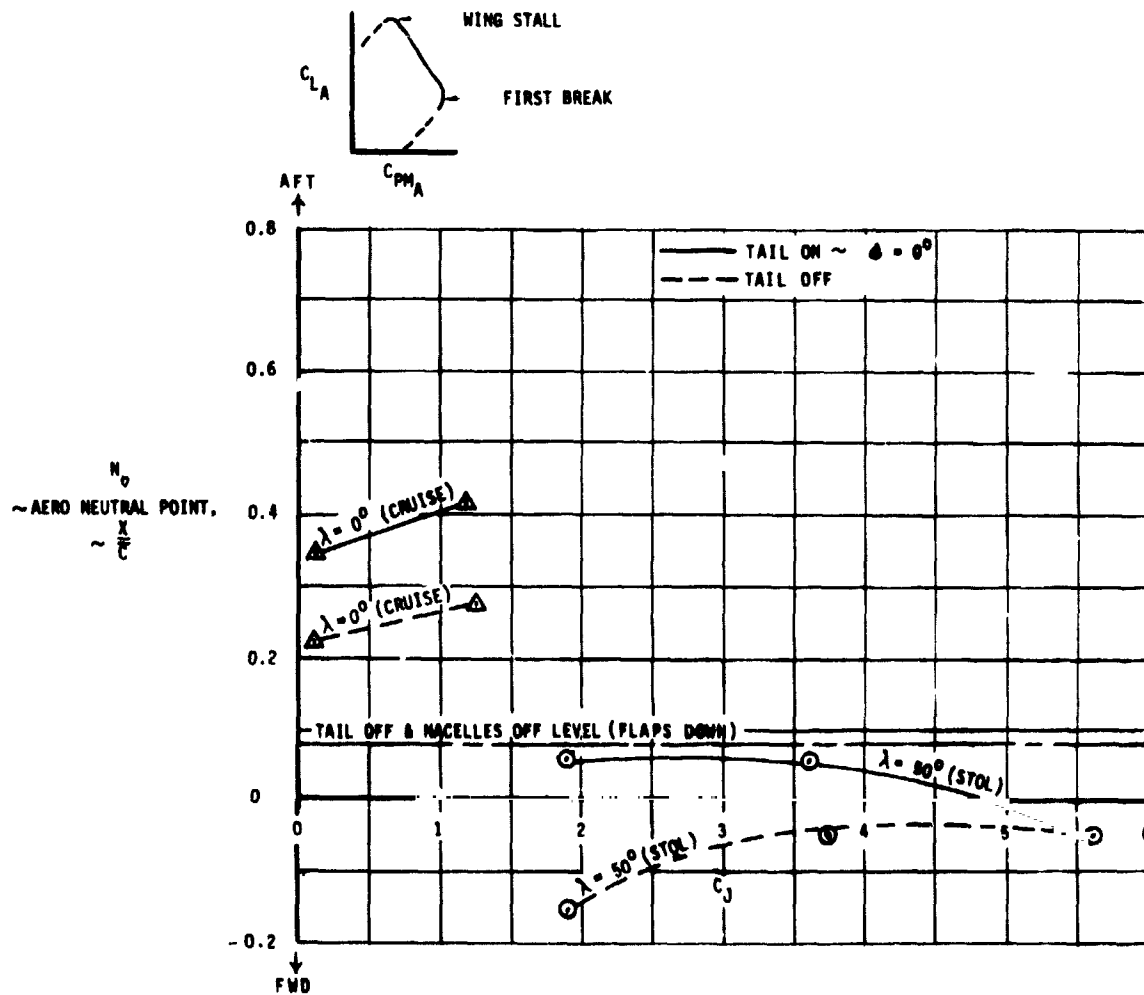


FIGURE 42 EFFECT OF THRUST ON NEUTRAL POINT LOCATION (HIGH C_{LA})

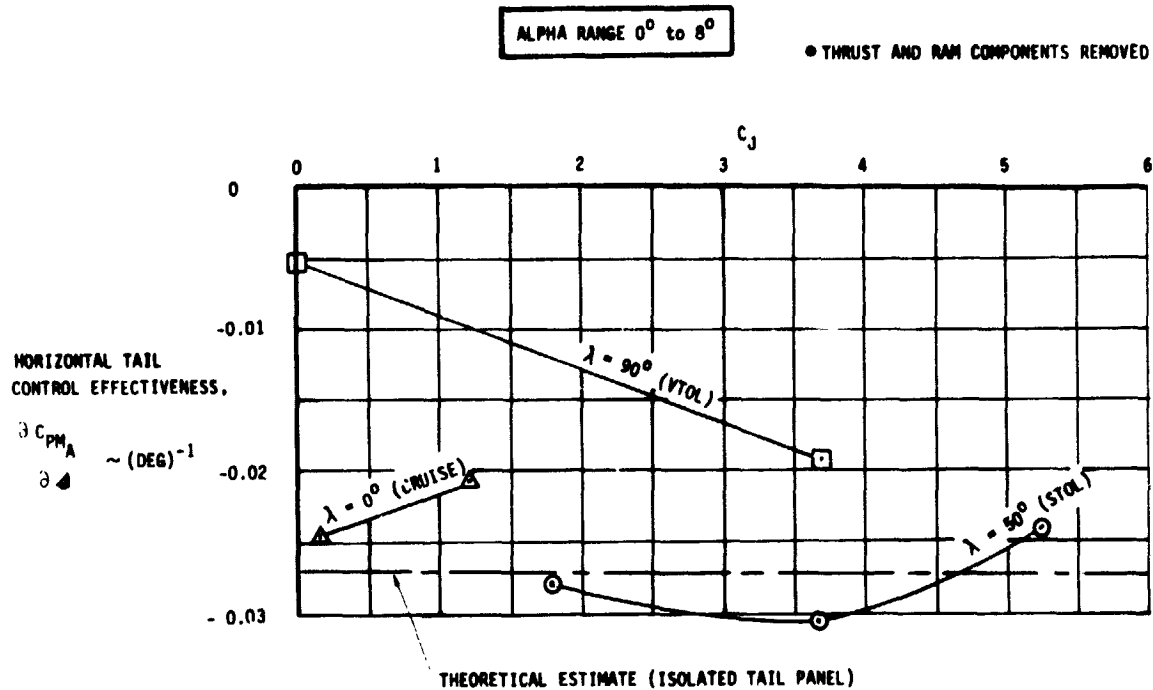


FIGURE 43 EFFECT OF THRUST ON HORIZONTAL TAIL CONTROL EFFECTIVENESS

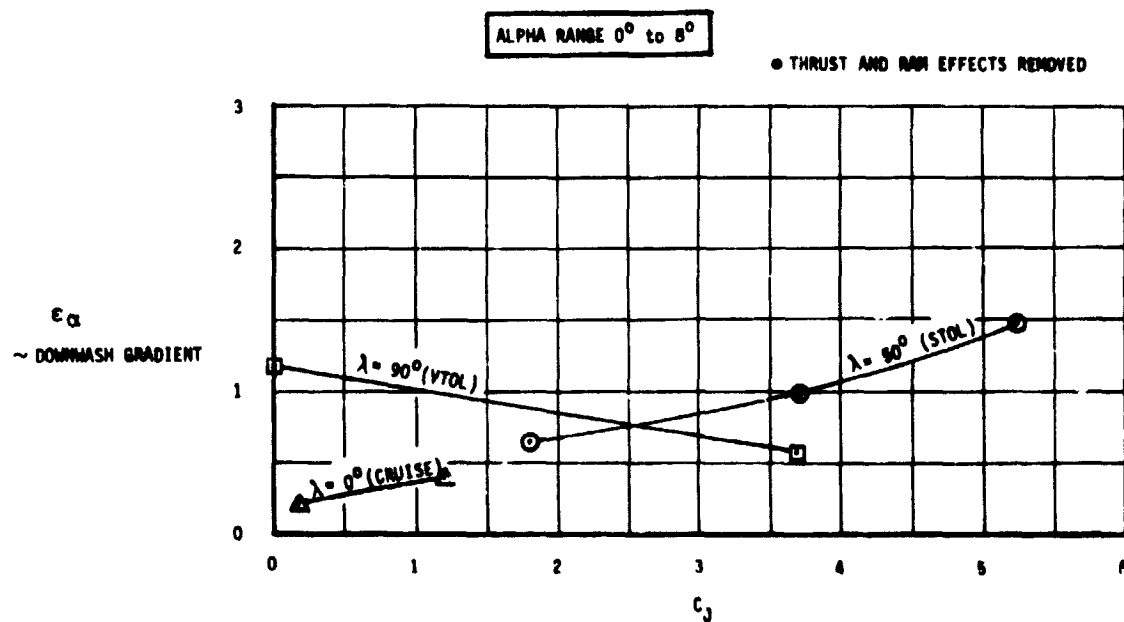


FIGURE 44 EFFECT OF THRUST ON DOWNWASH GRADIENT

- $\lambda = 90^\circ$
- FLAPS DOWN
- GEAR DOWN
- THRUST AND RAM COMPONENTS REMOVED
- $C_J = 3.6$
- $V/V_J = 0.29$

V.H. OFF DENOTES VERTICAL AND HORIZONTAL TAILS OFF

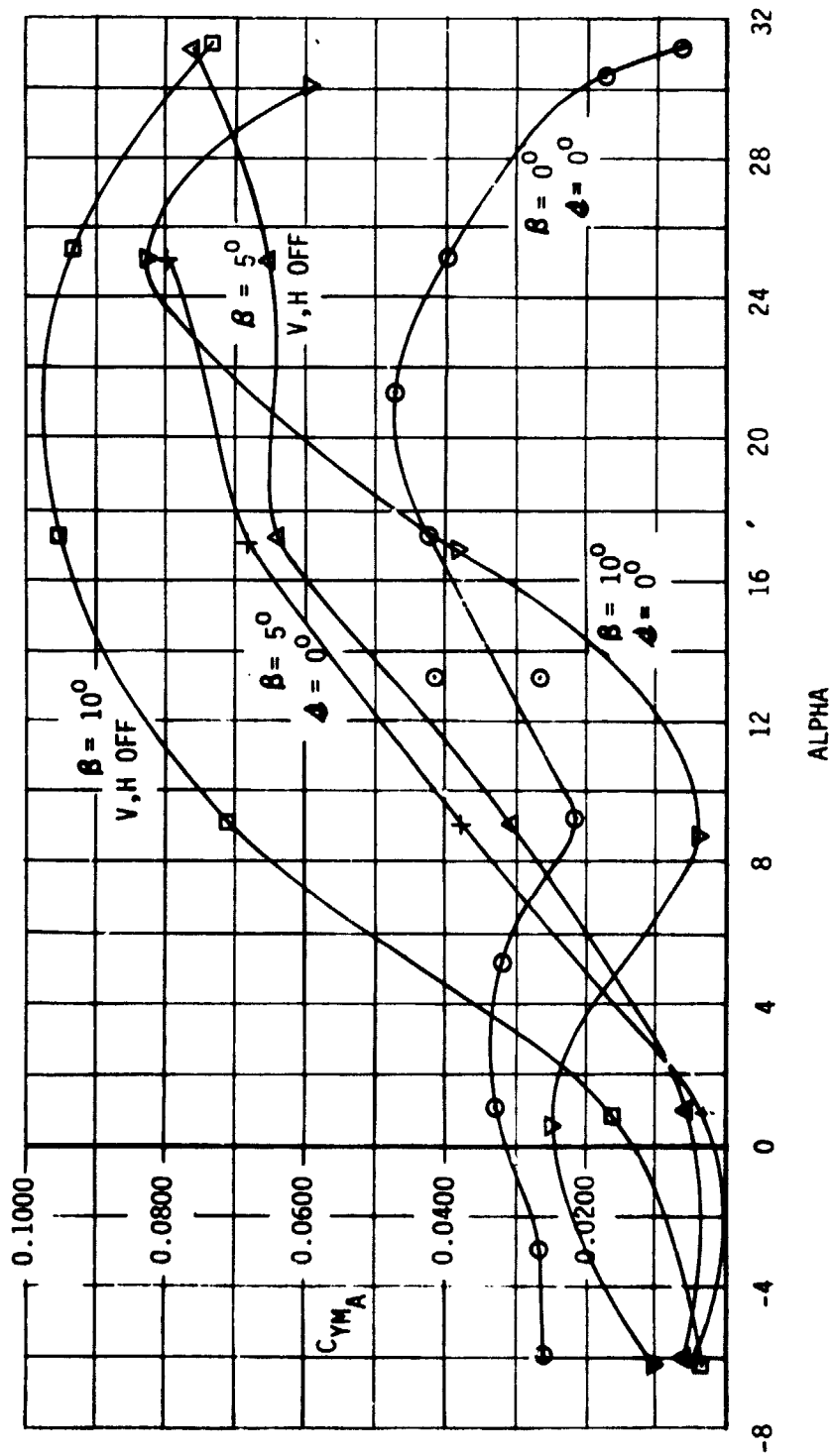


FIGURE 45 VERTICAL TAIL EFFECTIVENESS $\lambda = 90^\circ$, $C_J = 3.7$

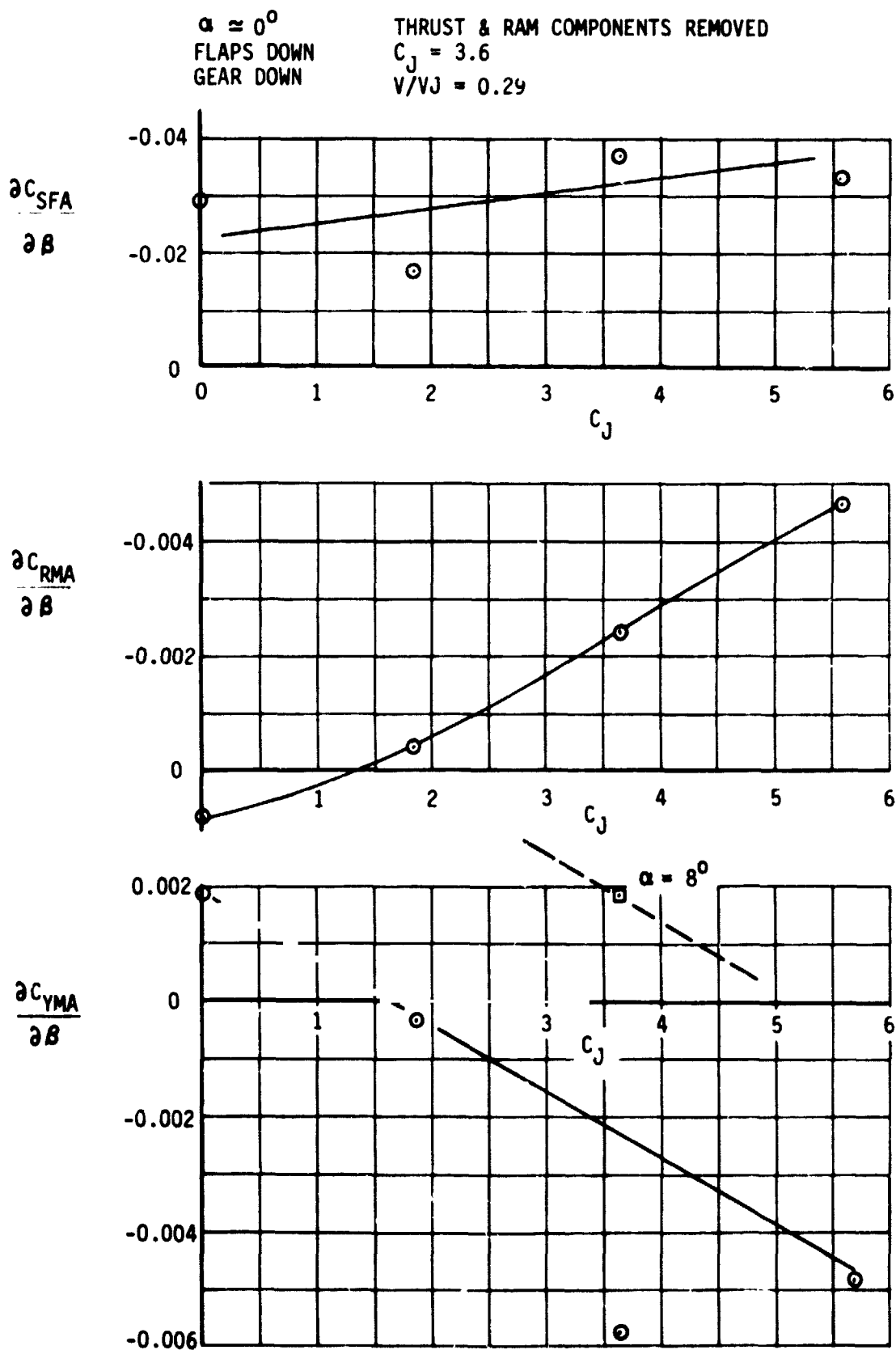


FIGURE 46 LATERAL - DIRECTIONAL DERIVATIVES $\lambda = 90^\circ$ LOW ANGLE OF ATTACK

— NACELLES ON, $\lambda = 0^\circ$, $C_J = \text{RAM}$

- - - NACELLES OFF

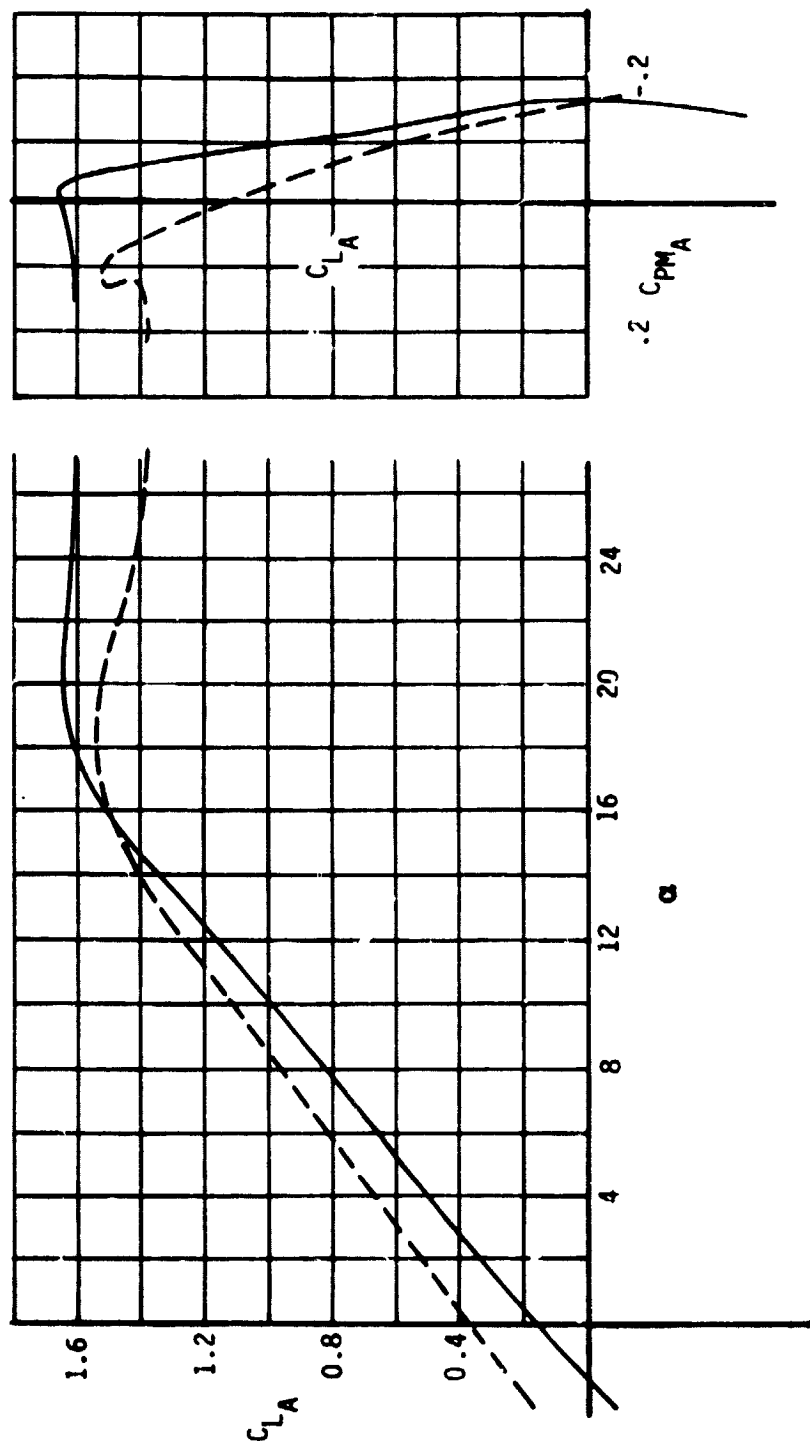


FIGURE 47 EFFECT OF NACELLES FLAPS DOWN

— NACELLES ON, $\lambda = 0$, $C_J = \text{RAM}$

- - - NACELLES OFF

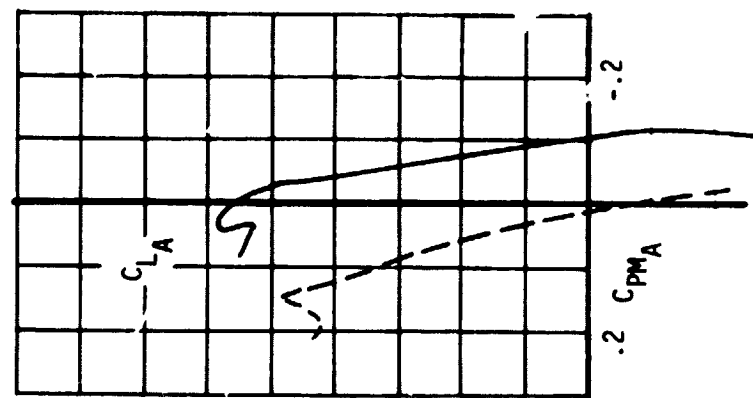
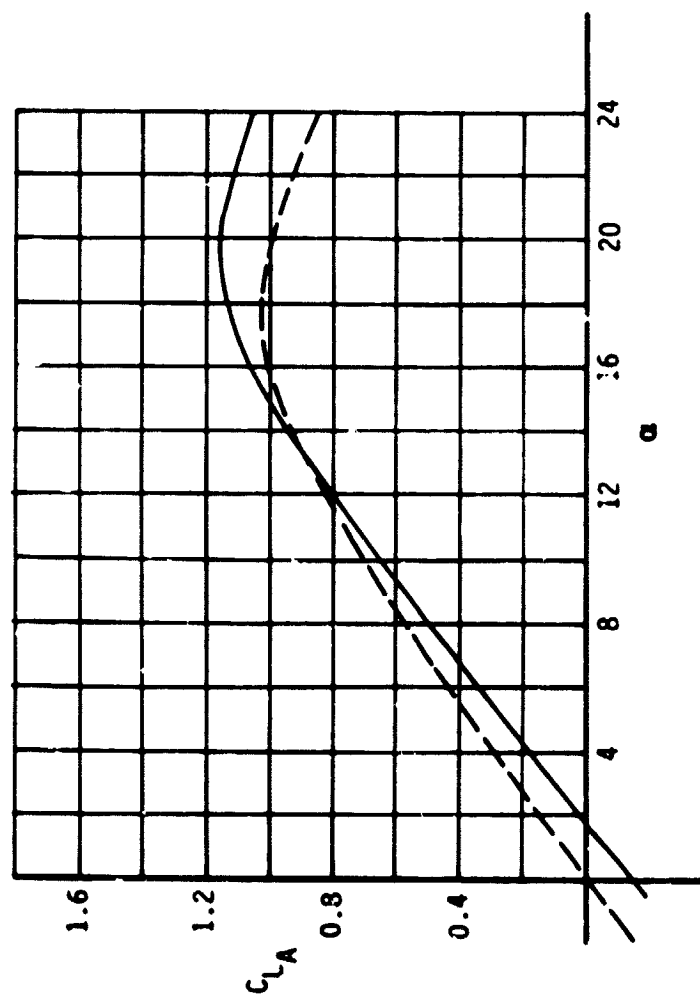


FIGURE 48 EFFECT OF NACELLES, FLAPS UP

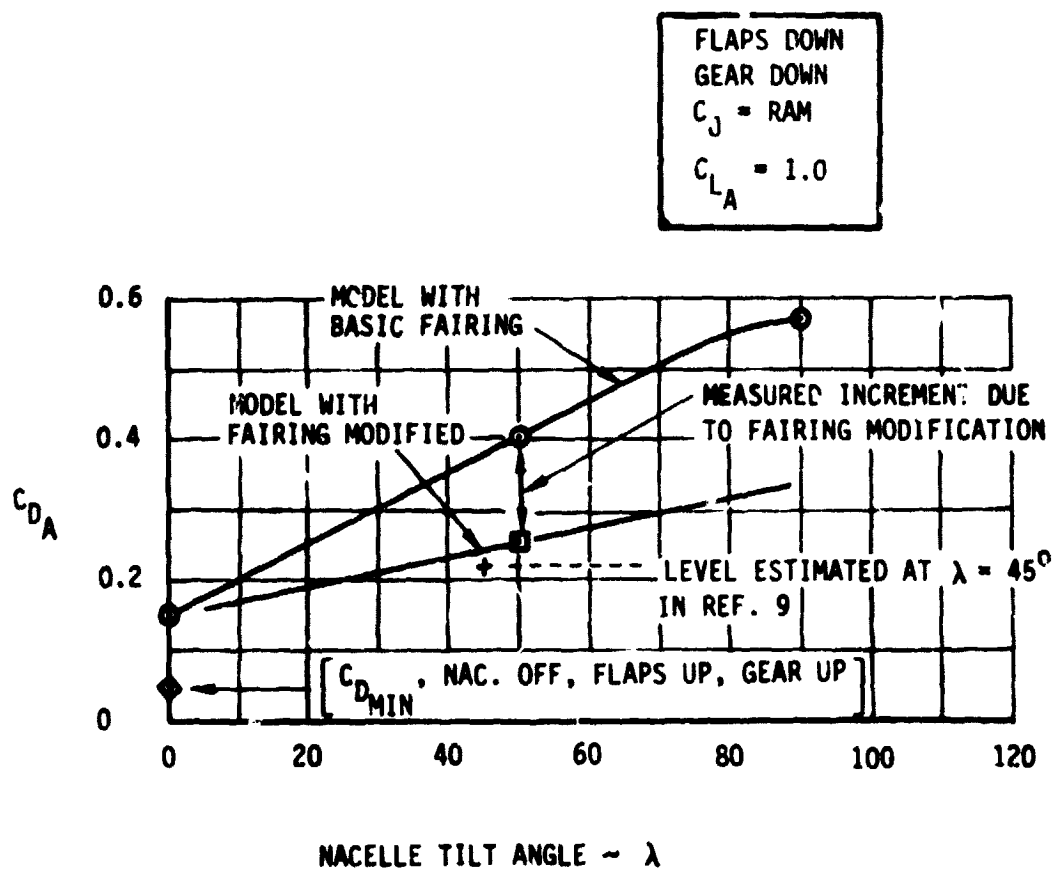


FIGURE 49 EFFECT OF NACELLES ON DRAG LEVEL, $C_J = \text{RAM}$

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APPENDIX A

WIND TUNNEL TEST DATA

INDEX OF FIGURES IN APPENDIX A

FIGURE	TITLE
A.1	VTOL configuration characteristics LAMBDA = 90^0 Tail Angle = 0^0
A.2	VTOL configuration characteristics LAMBDA = 90^0 H. Tail off
A.3	Effect of horizontal tail LAMBDA = 90^0 , $C_J = 0$
A.4	Effect of horizontal tail LAMBDA = 90^0 , $C_J = 3.7$
A.5	Effect of landing gear LAMBDA = 90^0 , $C_J = 3.7$
A.6	Effect of thrust level at $C_J \approx \text{CONST.}$
A.7	Effect of differential thrust LAMBDA = 90^0 , H. Tail off, $\beta = 0^0$
A.8	Effect of differential thrust LAMBDA = 90^0 , $C_J = 3.7$, $\beta = 5^0$
A.9	Effect of differential thrust LAMBDA = 90^0 , $C_J = 3.7$, $\beta = 10^0$
A.10	Vertical tail effectiveness LAMBDA = 90^0 , $C_J = 0$
A.11	Vertical tail effectiveness LAMBDA = 90^0 , $C_J = 0$
A.12	Vertical tail effectiveness LAMBDA = 90^0 , $C_J = 1.8$
A.13	Vertical tail effectiveness LAMBDA = 90^0 , $C_J = 1.8$
A.14	Vertical tail effectiveness LAMBDA = 90^0 , $C_J = 3.7$
A.15	Vertical tail effectiveness LAMBDA = 90^0 , $C_J = 3.7$
A.15	Vertical tail effectiveness LAMBDA = 90^0 , $C_J = 5.5$

FIGURE	TITLE
A.17	Vertical tail effectiveness LAMBDA = 90° , $C_J = 5.5$
A.18	Effect of engine tilt angle $C_J = 3.7$ ground plane installed
A.19	Effect of engine tilt angle $C_J = 5.5$, ground plane installed
A.20	Effect of ground plane LAMBDA = 90° , $C_J = 3.7$
A.21	Effect of ground plane LAMBDA = 50° , $C_J = 1.8$
A.22	STOL configuration characteristics LAMBDA = 50° , H. Tail off
A.23	STOL configuration characteristics LAMBDA = 50° , tail angle = 10°
A.24	Effect of horizontal tail LAMBDA = 50° , $C_J = 1.8$
A.25	Effect of horizontal tail LAMBDA = 50° , $C_J = 3.7$
A.26	Effect of nose-fan thrust variation LAMBDA = 50° , tail angle = 0°
A.27	Effect of nose fan thrust variation LAMBDA = 50° , H. Tail off
A.28	Effect of Aileron Droop LAMBDA = 50°
A.29	Effect of landing gear LAMBDA = 50°
A.30	CTOL configuration characteristics LAMBDA = 0° , H. Tail off
A.31	Effect of landing gear LAMBDA = 0° , flaps down, power off
A.32	Effect of landing gear LAMBDA = 0° , flaps down, power on
A.33	Effect of flap and slat LAMBDA = 0°

FIGURE	TITLE
A.34	Effect of horizontal tail LAMBDA = 0^0 , flaps up, $C_J = 1.2$
A.35	Effect of horizontal tail LAMBDA = 0^0 , flaps up, $C_J = 1.8$
A.36	Effect of flap door Nacelles off
A.37	Effect of flap and slat Nacelles off
A.38	Configuration buildup Power off
A.39	Effect of nose fan thrust variation LAMBDA = 0^0 , H. Tail off

APPENDIX A

WIND TUNNEL TEST DATA

Appendix A contains plots of all valid data runs obtained from the wind tunnel tests conducted under this contract. Each figure consists of three pages for symmetrical data, or six pages for asymmetrical data. Parameters plotted are:

Symmetrical Data	{ CLA vs ALPHA CDA vs CLA CPMA vs CLA
Asymmetrical Data	{ The above parameters plus CRMA vs ALPHA CYMA vs ALPHA CSFA vs ALPHA

The data represent "aerodynamic data" in that direct thrust and ram forces have been removed. The aerodynamic data does contain propulsion induced effects. The determination of direct thrust and ram forces is discussed in Appendix B and the removal of direct thrust and ram forces is discussed in Appendix C. Also presented in Appendix C (Figure C.1) is a tracking of data through the reduction process and a description of the wind tunnel wall constraint corrections which were applied to the data.

Out of necessity, a portion of the lift/cruise fan nacelle strut was rotated with the nacelles during the wind tunnel test program to cover plumbing to the turbopowered simulators and the "blown" primary. That fairing had a large effect upon the drags of the STOL and VTOL test configurations which would not be present on the full scale airplane. To assess the drag difference due to the fairing, a test run was made with the fairing removed and tape and wax applied to cover and fair

over the exposed plumbing and instrumentation. As discussed in the body of the report, the reduction in drag coefficient due to removing the fairing amounted to 0.15 but further reductions are probably possible. The drag data in this appendix are not representative of airplane drag levels to the extent that they are affected by this fairing.

Figures A.18 thru A.21 contain data which were taken with a ground plane installed in the tunnel. While these data do not represent ground height series, there was some variation of ground height as the model was pitched. Since these data were corrected for wall effects of three walls only, the ground effect is a part of the data. The ground height to model mean aerodynamic chord ratios can be represented by straight lines between the following three points: $\alpha = -5^\circ$, $h/\bar{c} = 2.45$; $\alpha = 12^\circ$, $h/\bar{c} = 3.40$; $\alpha = 31^\circ$, $h/\bar{c} = 4.90$. It should also be noted that the ground plane reduced the tunnel width (which is "height" relative to the model since it is installed with wings vertical) from 105 inches to 77 inches.

Model Nomenclature

The plots in this appendix contain a shorthand notation to designate the specific model configuration. This notation is defined as follows:

- C1 Trailing edge flaps deflected
 Trailing edge flap door down
 Leading edge slat extended
 Nose fan doors in open position (both upper & lower)
 Basic nacelles and nacelle - fuselage fairings installed

- C2 Same as C1 except trailing edge flap door is up
 (i.e., it is aligned with the basic trailing
 edge flap)

- C3 Trailing edge flaps deflected
 Trailing edge flap door up
 Leading edge slat extended
 Nose fan doors in closed position (both upper & lower)
 Basic nacelles and nacelle - fuselage fairings installed

- C4 Trailing edge flaps & flap door nested
 Leading edge slat nested
 Nose fan doors in closed position (both upper & lower)
 Basic nacelles and nacelle - fuselage fairings installed

- C5 The lift/cruise nacelles and the portion of the nacelle -
 fuselage fairing attached to the nacelles are removed from
 the model. The resulting cavities on the body sides are
 taped over for a smooth fairing.
 Nose fan doors in closed position
 Leading edge slat nested
 Trailing edge flap position as noted by "F" designation

- G Main and nose landing gear and landing gear doors installed

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V Vertical tail installed

H Horizontal tail installed

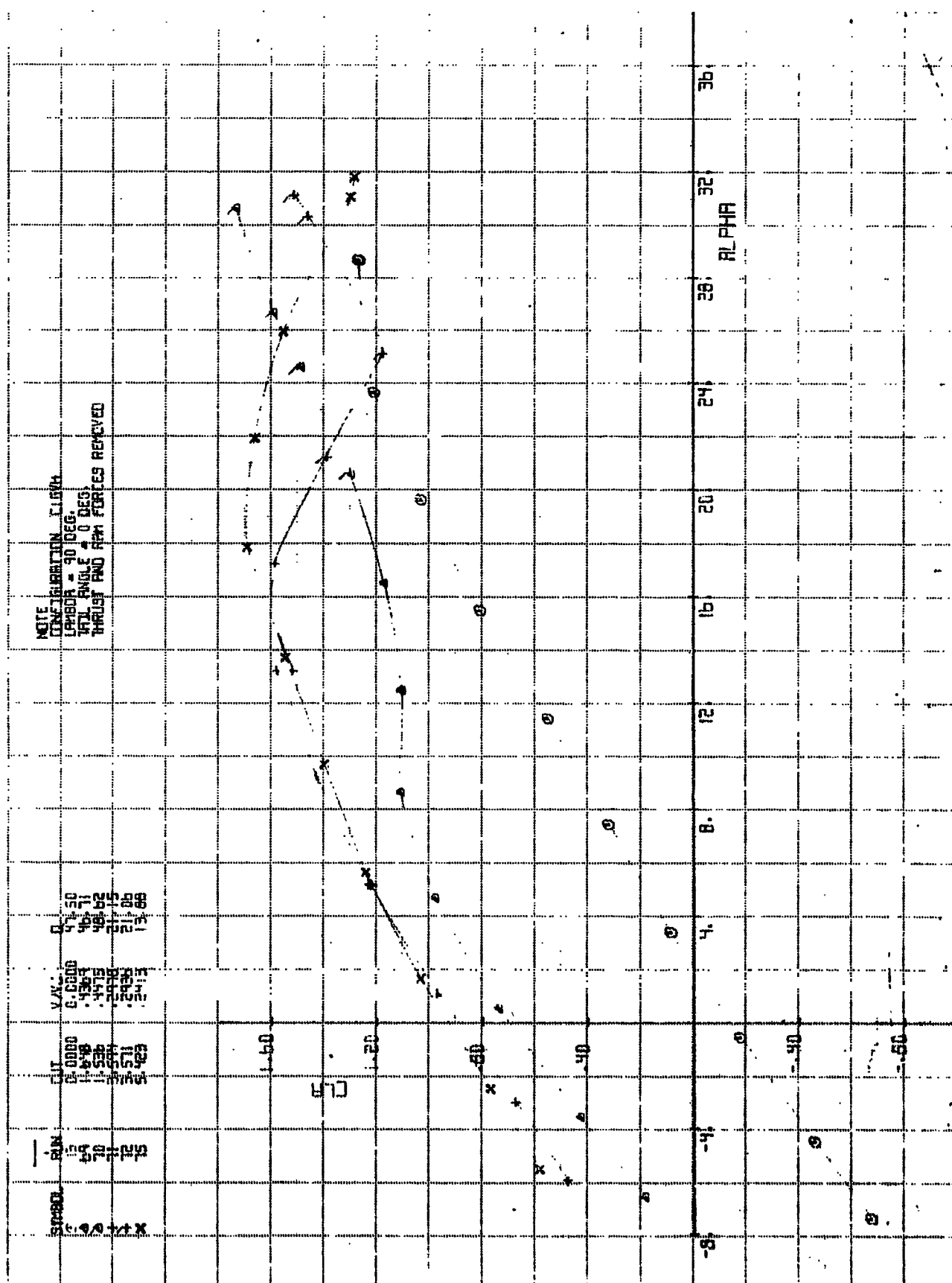
F25/25 Trailing edge flap deflected
Trailing edge flap door aligned with flap

F25/110 Trailing edge flap deflected
Trailing edge flap door down

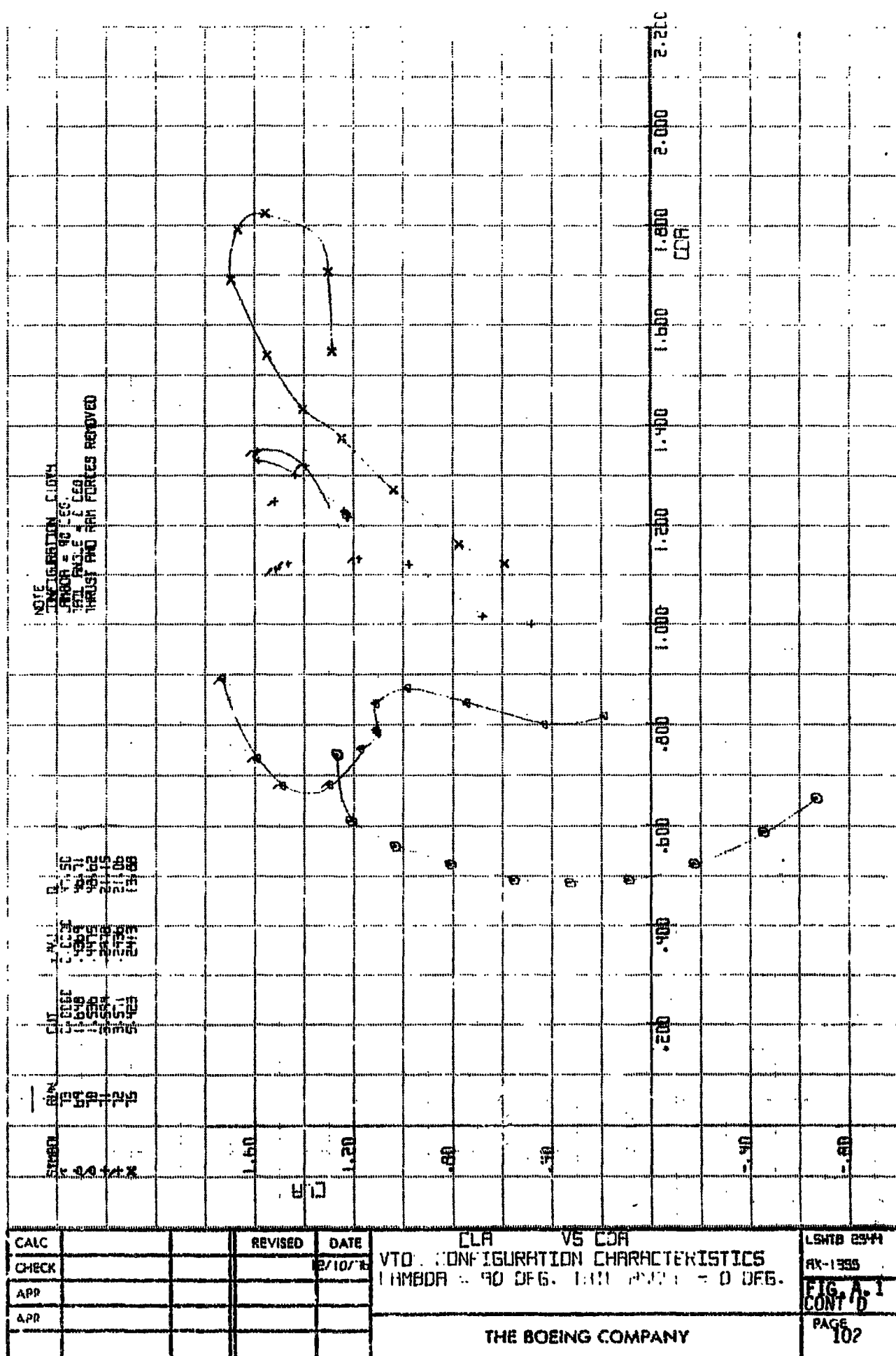
Note: C1 thru C5 do not have tail surfaces and gear installed
unless followed by the appropriate symbols,

Example: C1GVH is C1 with gear, vertical tail and horizontal tail
installed.

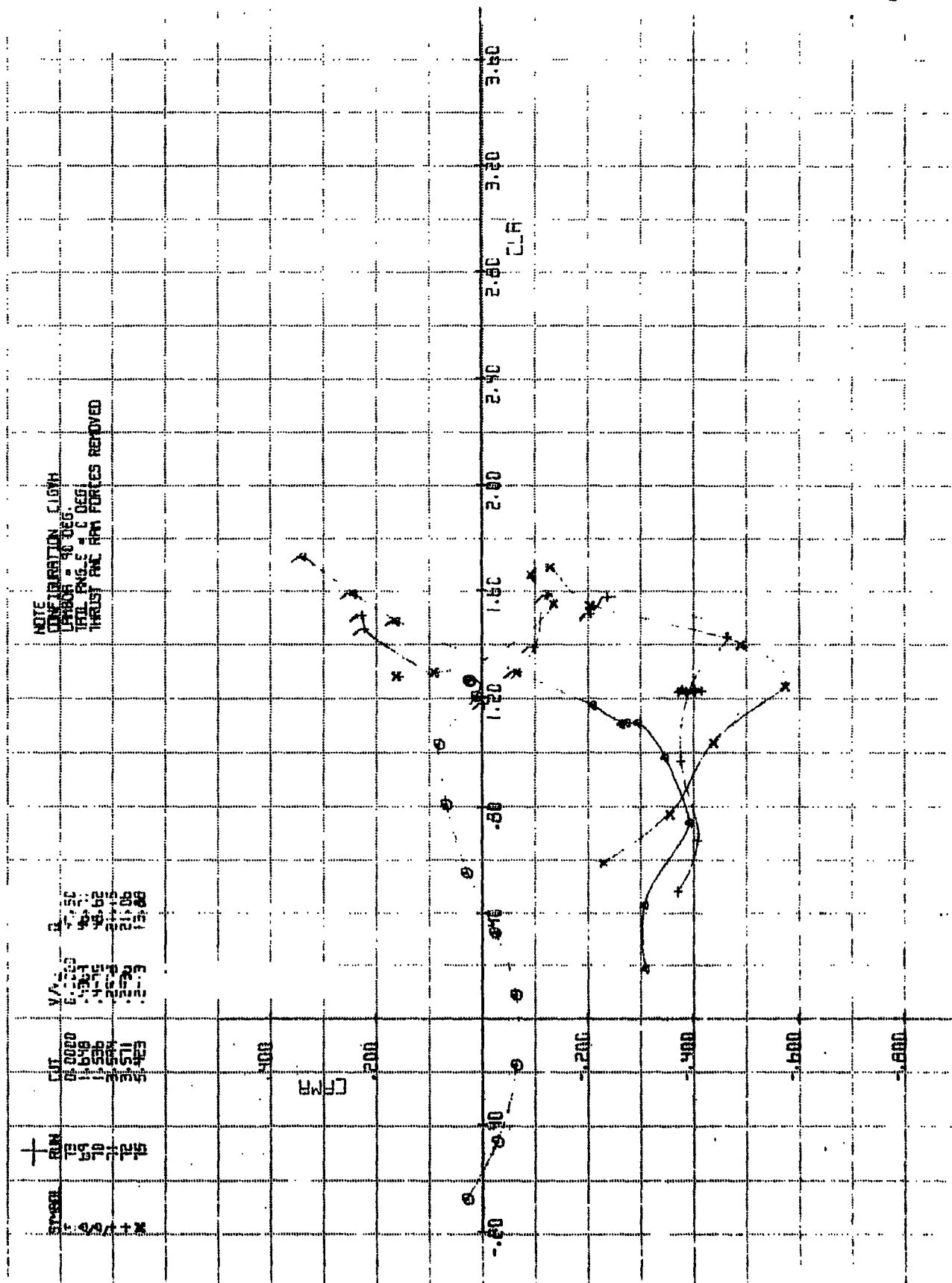
NOTE
CONFIGURATION CUSH
LAMBDA = 90 DEG.
TAIL ANGLE = 0 DEG.
THRUST AND RPM FORCES REMOVED



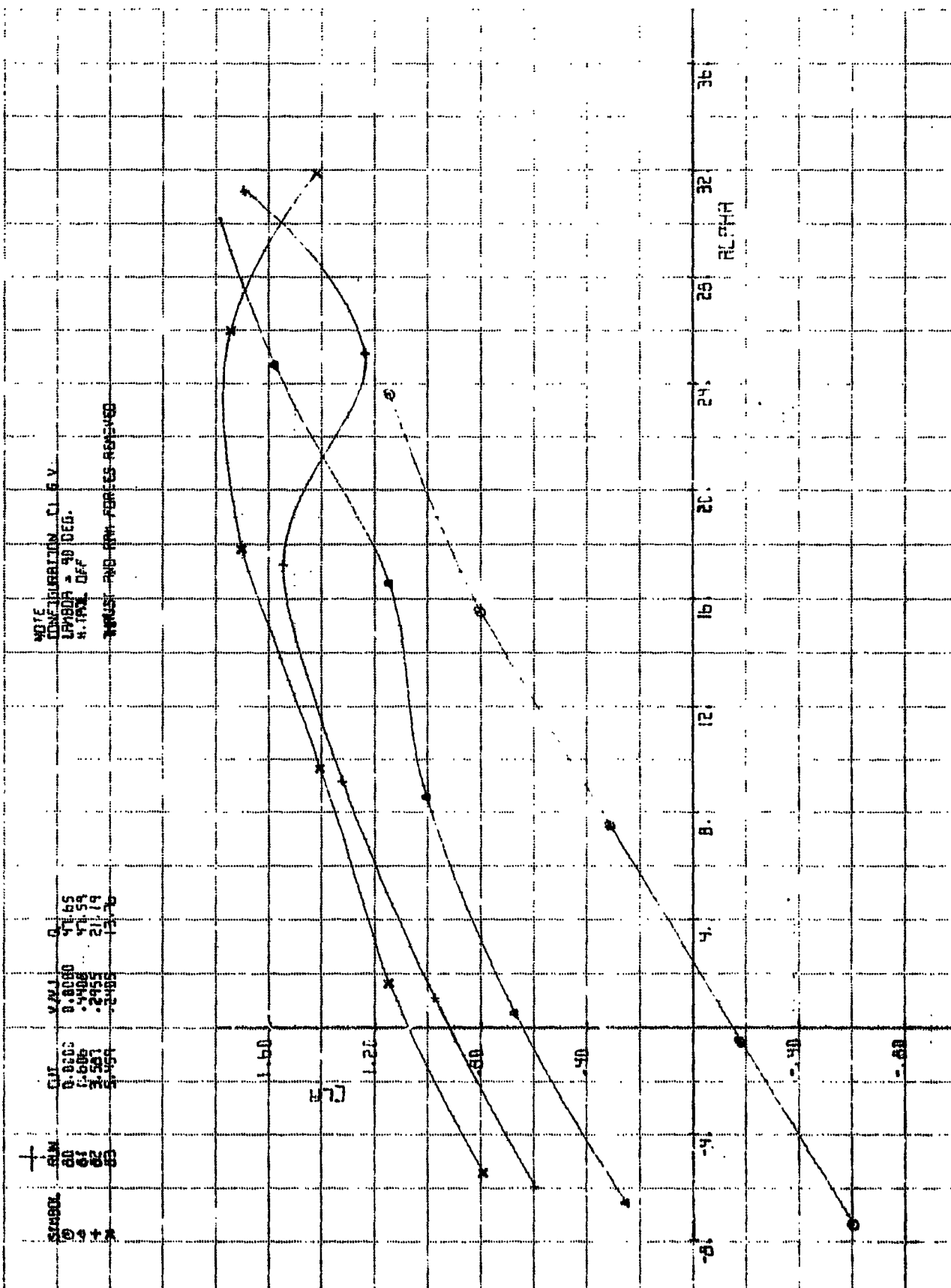
CALC			REVISED	DATE	ETA VS ALPHA	LSWTS 2541
CHECK				2/10/78	VTL CONFIGURATION CHARACTERISTICS	AX-1355
APR					LAMBDA = 90 DEG. TAIL ANGLE = 0 DEG.	FIG. A.1
APR					THE BOEING COMPANY	PAGE 101



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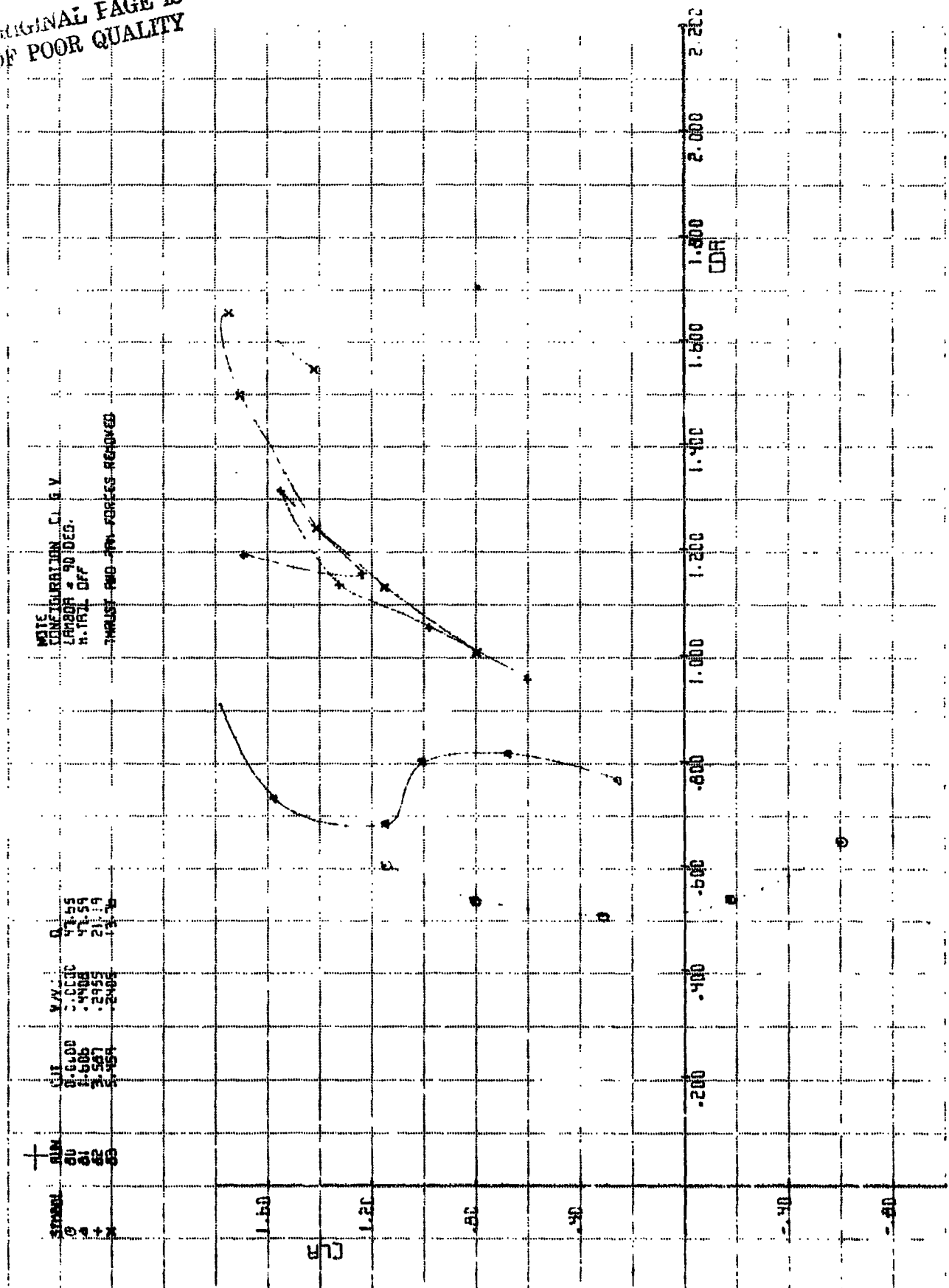
CALC		REVISED	DATE	CPMA VS C/L	LRWTD 2511
CHECK			12/10/16	VTOL CONFIGURATION CHARACTERISTICS	AX-1355
APR				LAMBDA = 90 DEG. TAIL ANGLE = 0 DEG.	FIG. A.1
APR					CONT'D
				THE BOEING COMPANY	PAGE 103



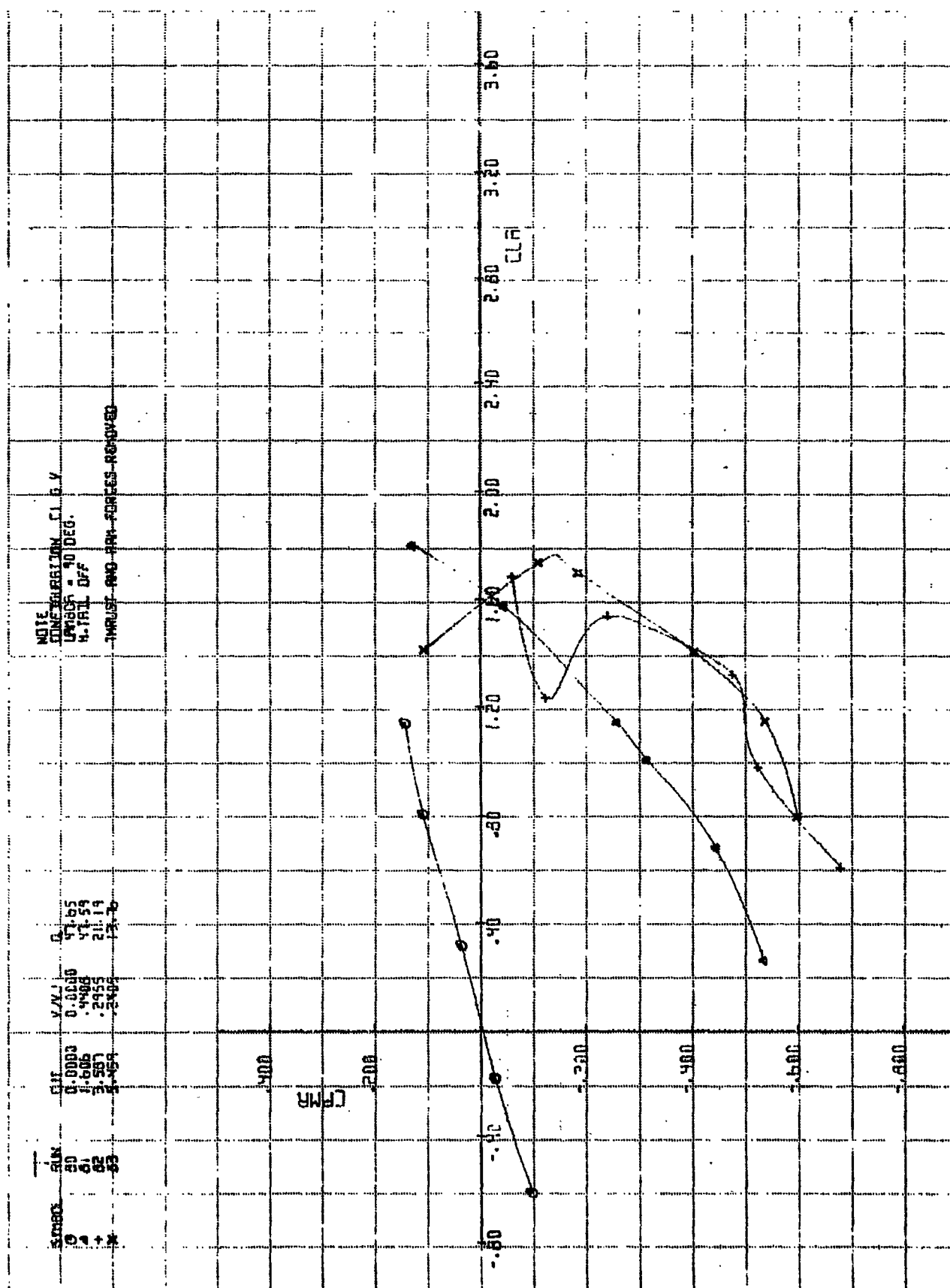
CL	VS ALPHA	LSMTB 25-4
VTOL CONFIGURATION CHARACTERISTICS		AX-135
LAMBDA = 90 DEG. H. TAIL OFF		FIG. A. 2
THE BOEING COMPANY		PAGE 104

CALC	REVISD	DATE
CHECK		01.12/71
APR		
APR		

Year	Age	Sex	Height	Weight	Arm span	Hand span	Hand width	Hand length	Hand area	Hand volume	Hand mass	Hand density
1990	20	M	170	65	180	18	8.5	10.5	88	1.2	1.2	1.2
1991	21	M	175	70	185	19	9.0	11.0	95	1.3	1.3	1.3
1992	22	M	180	75	190	20	9.5	11.5	100	1.4	1.4	1.4
1993	23	M	185	80	195	21	10.0	12.0	105	1.5	1.5	1.5
1994	24	M	190	85	200	22	10.5	12.5	110	1.6	1.6	1.6
1995	25	M	195	90	205	23	11.0	13.0	115	1.7	1.7	1.7
1996	26	M	200	95	210	24	11.5	13.5	120	1.8	1.8	1.8
1997	27	M	205	100	215	25	12.0	14.0	125	1.9	1.9	1.9
1998	28	M	210	105	220	26	12.5	14.5	130	2.0	2.0	2.0
1999	29	M	215	110	225	27	13.0	15.0	135	2.1	2.1	2.1
2000	30	M	220	115	230	28	13.5	15.5	140	2.2	2.2	2.2
2001	31	M	225	120	235	29	14.0	16.0	145	2.3	2.3	2.3
2002	32	M	230	125	240	30	14.5	16.5	150	2.4	2.4	2.4
2003	33	M	235	130	245	31	15.0	17.0	155	2.5	2.5	2.5
2004	34	M	240	135	250	32	15.5	17.5	160	2.6	2.6	2.6
2005	35	M	245	140	255	33	16.0	18.0	165	2.7	2.7	2.7
2006	36	M	250	145	260	34	16.5	18.5	170	2.8	2.8	2.8
2007	37	M	255	150	265	35	17.0	19.0	175	2.9	2.9	2.9
2008	38	M	260	155	270	36	17.5	19.5	180	3.0	3.0	3.0
2009	39	M	265	160	275	37	18.0	20.0	185	3.1	3.1	3.1
2010	40	M	270	165	280	38	18.5	20.5	190	3.2	3.2	3.2
2011	41	M	275	170	285	39	19.0	21.0	195	3.3	3.3	3.3
2012	42	M	280	175	290	40	19.5	21.5	200	3.4	3.4	3.4
2013	43	M	285	180	295	41	20.0	22.0	205	3.5	3.5	3.5
2014	44	M	290	185	300	42	20.5	22.5	210	3.6	3.6	3.6
2015	45	M	295	190	305	43	21.0	23.0	215	3.7	3.7	3.7
2016	46	M	300	195	310	44	21.5	23.5	220	3.8	3.8	3.8
2017	47	M	305	200	315	45	22.0	24.0	225	3.9	3.9	3.9
2018	48	M	310	205	320	46	22.5	24.5	230	4.0	4.0	4.0
2019	49	M	315	210	325	47	23.0	25.0	235	4.1	4.1	4.1
2020	50	M	320	215	330	48	23.5	25.5	240	4.2	4.2	4.2
2021	51	M	325	220	335	49	24.0	26.0	245	4.3	4.3	4.3
2022	52	M	330	225	340	50	24.5	26.5				

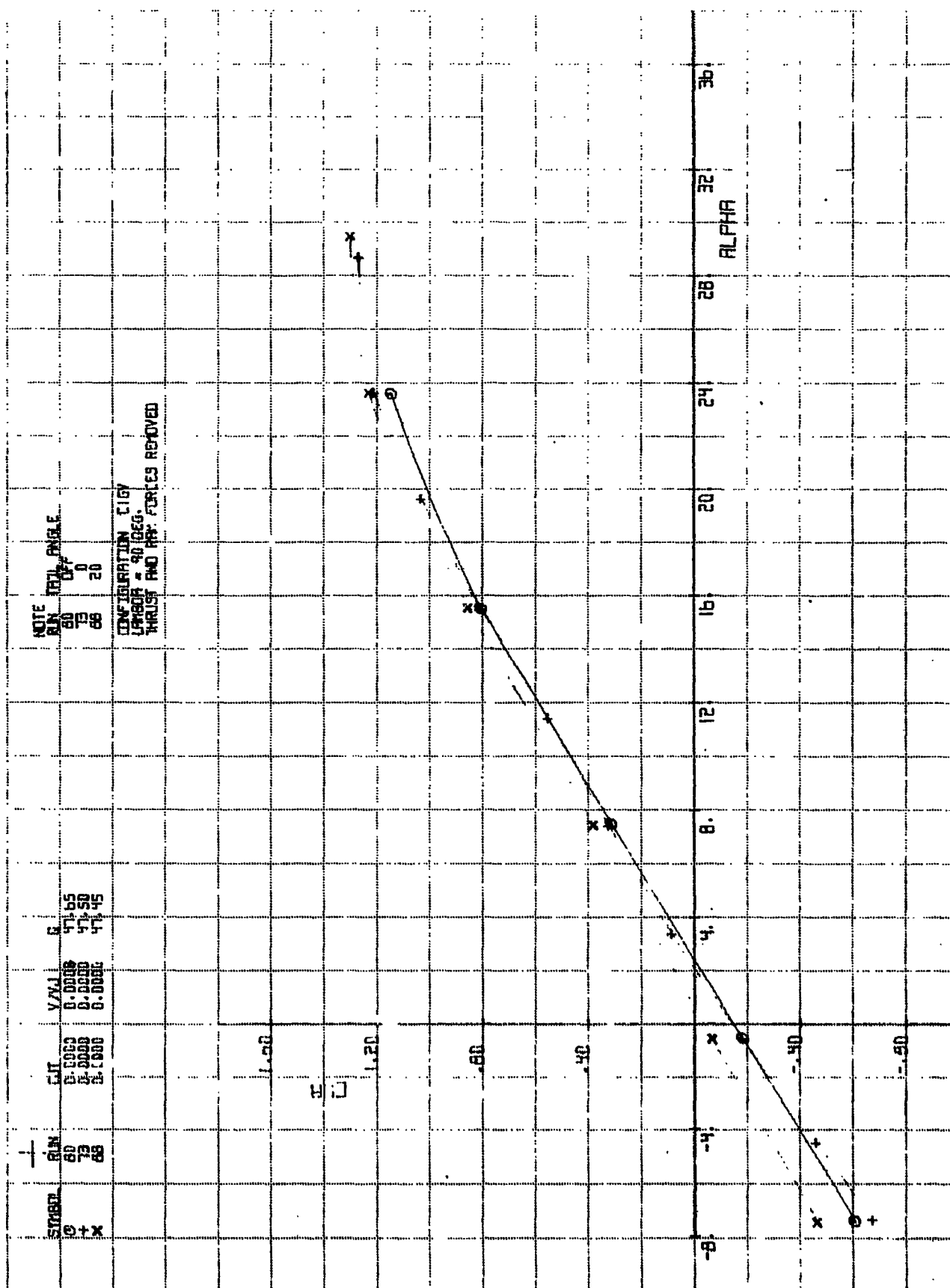


CALC			REVISED	DATE	CIA VS CDR VTOL CONFIGURATION CHARACTERISTICS LAMBDA = 90 DEG. H.TAIL OFF	LSWIB 2571 AX-135J FIG. A.2 CONT'D PAGE 105
CHECK				01/12/71		
APR						
APR						
					THE BOEING COMPANY	



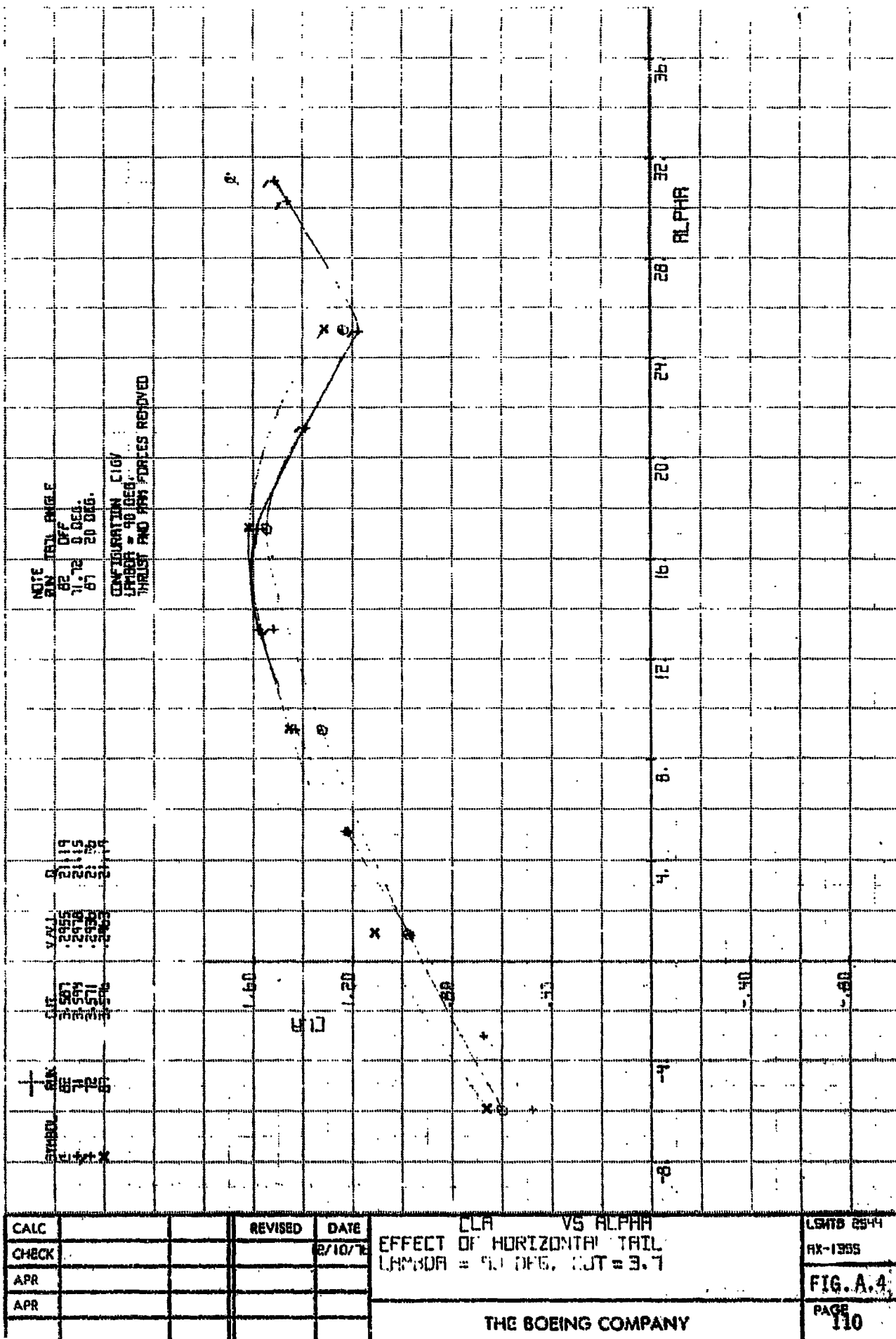
CLC		REVISED	DATE	CLM VS CLF	LGHTS 25-H
CHECK			01/12/77	VTOL CONFIGURATION CHARACTERISTICS	RX-1355
APR				LAMBDA = 90 DEG. H. TAIL OFF	FIG. A.2
APR					CONT'D
				THE BOEING COMPANY	PAGE 106

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CLC		REVISED	DATE	CLT VS ALPHA	LSMTB 2544
CHECK			12/10/76	EFFECT OF HORIZONTAL TAIL	AX-1355
				LAMBDA = 90 DEG., CLT = 0	FIG. A.3
				THE BOEING COMPANY	PAGE 107





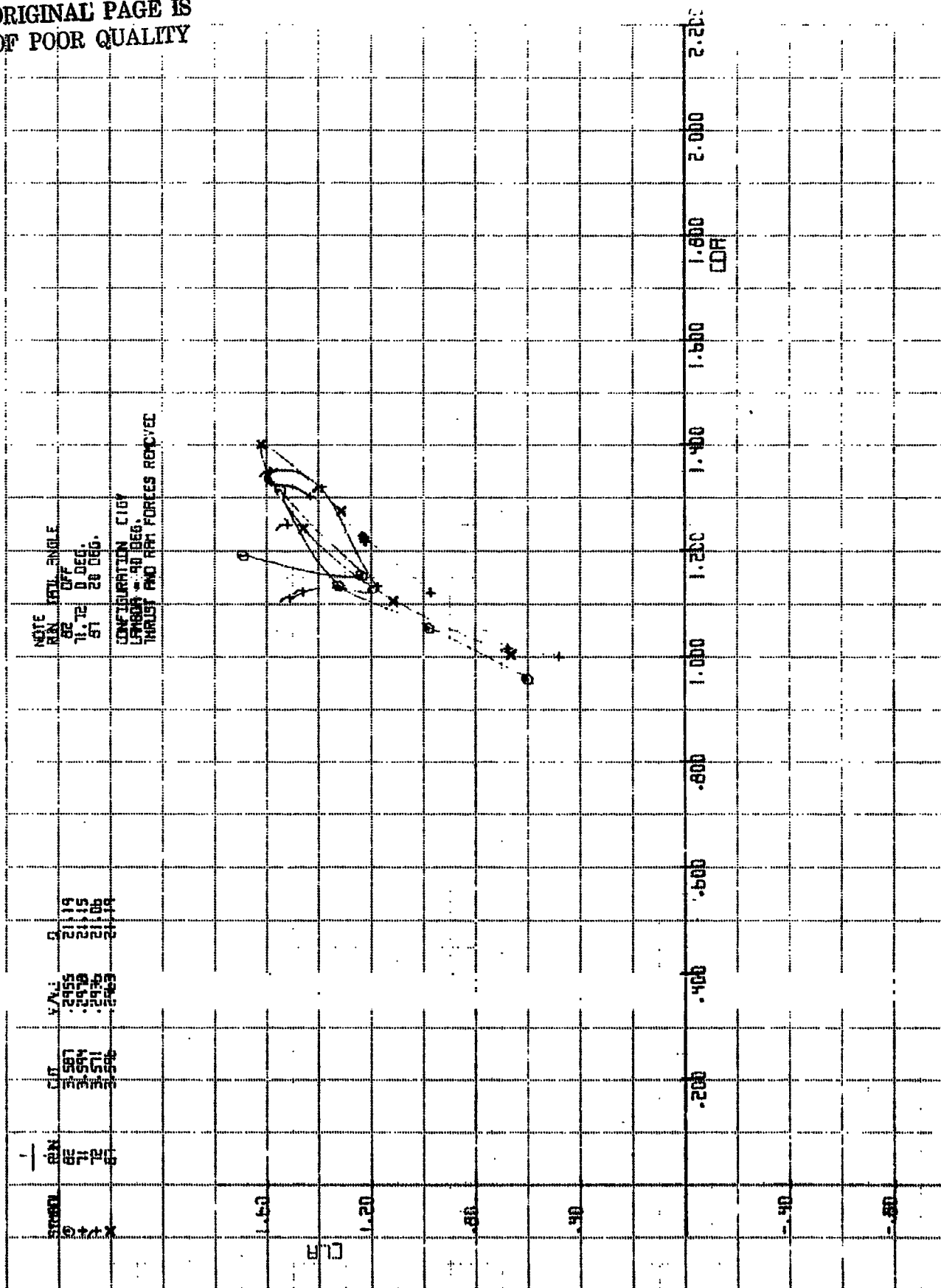
CALC		REVISED	DATE
CHECK			2/10/76
APR			
APR			

CL vs ALPHA
EFFECT OF HORIZONTAL TAIL
LAMBDA = 90 DEG, NOT = 3.7

LSWTS 2544
RX-1355

FIG. A.4

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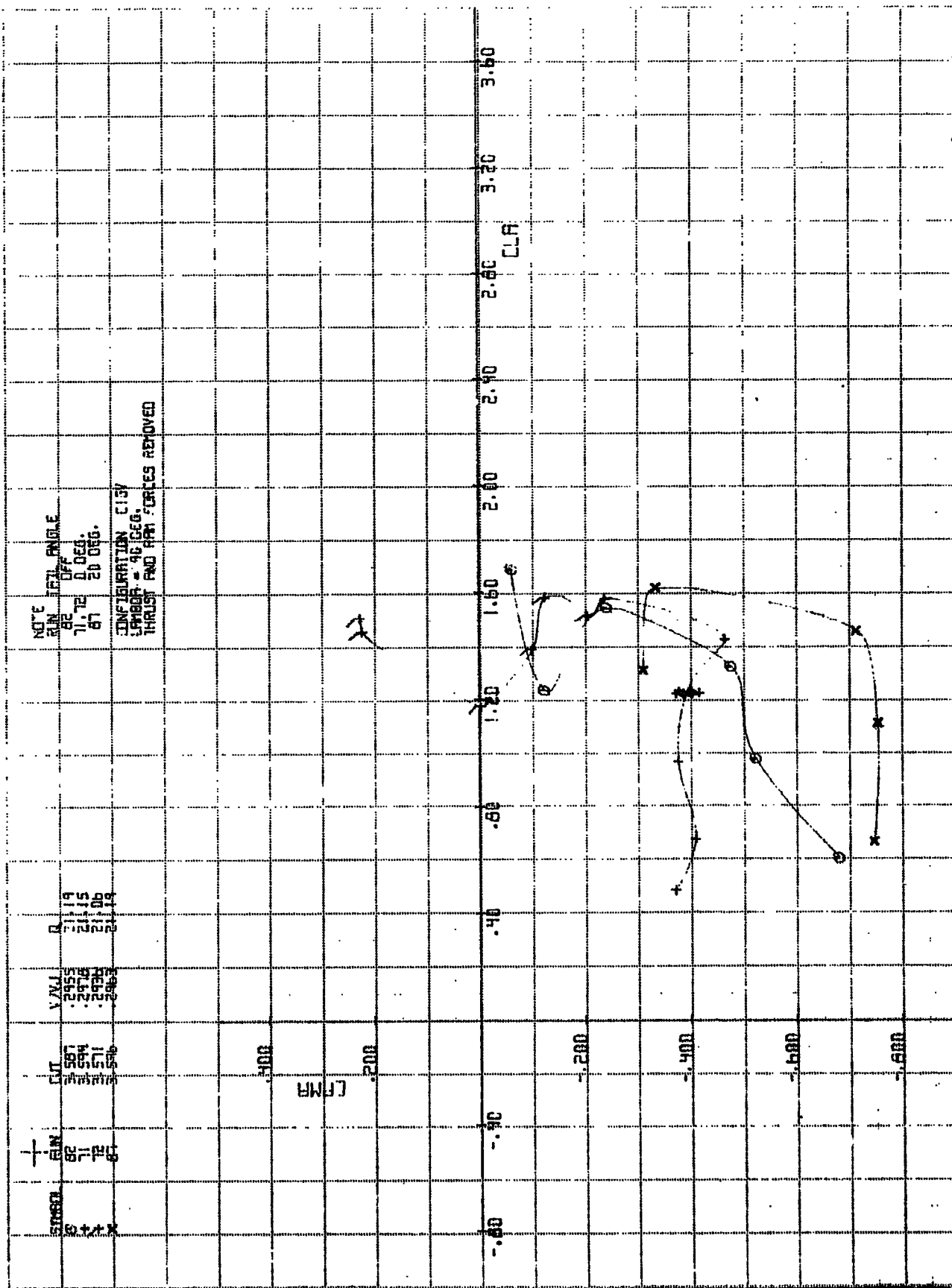
NOTE
REAL TAIL ANGLE
82 OFF
11.12 0 DEG.
57 28 DEG.

CONFIGURATION C16
LAMBDA = 90 DEG.
THRUST AND AIR FORCES REMOVED

CG
0.00
0.10
0.20
0.30
0.40
0.50
0.60
0.70
0.80
0.90
1.00
1.10
1.20
1.30
1.40
1.50
1.60
1.70
1.80
1.90
2.00
2.10
2.20

CG
0.00
0.10
0.20
0.30
0.40
0.50
0.60
0.70
0.80
0.90
1.00
1.10
1.20
1.30
1.40
1.50
1.60
1.70
1.80
1.90
2.00
2.10
2.20

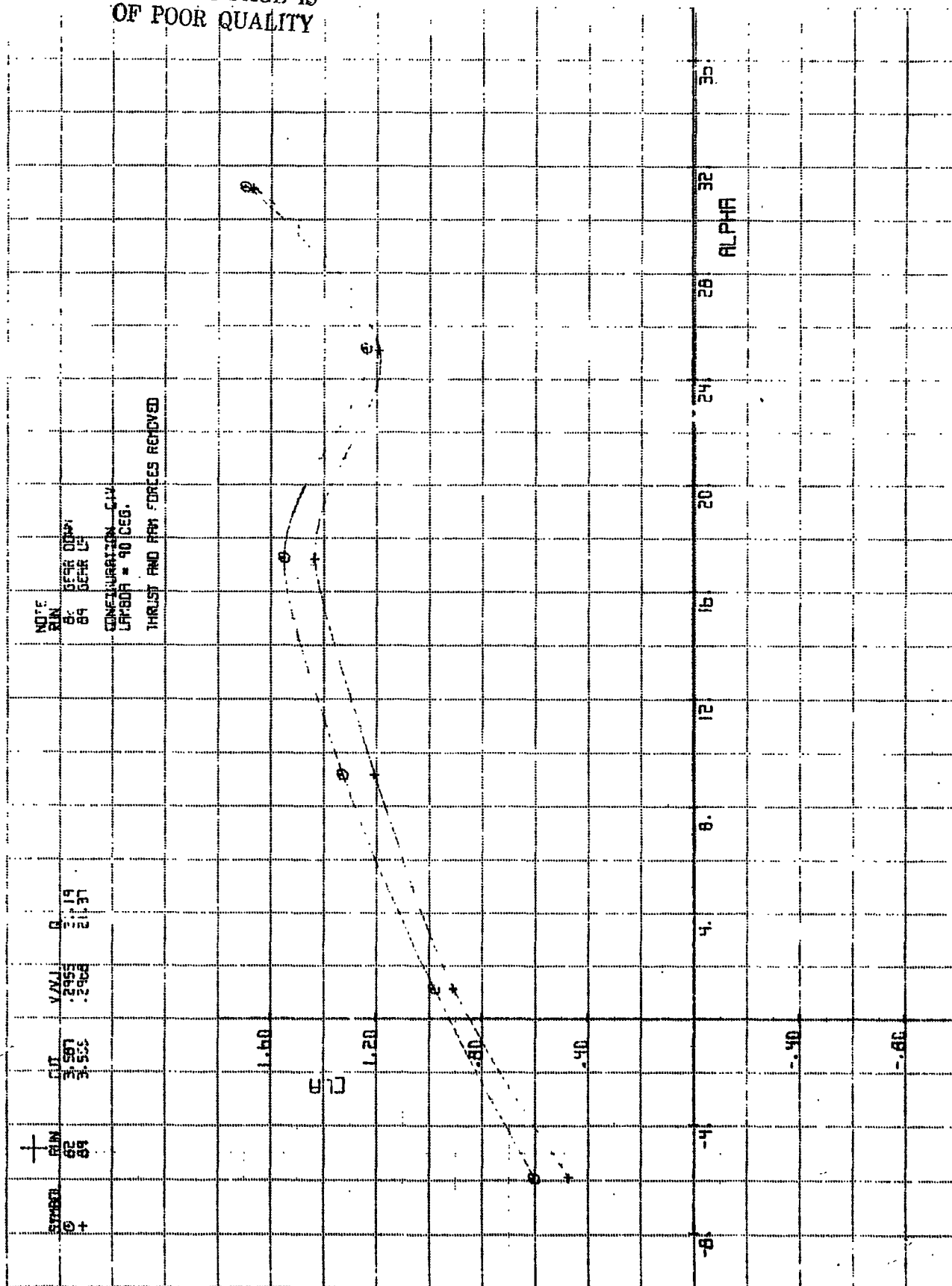
CALC		REVISED	DATE	CLA VS CGA	LEMBTB 25+1
CHECK			2/10/72	EFFECT OF HORIZONTAL TAIL	AX-1355
APR				LAMBDA = 90 DEG. CJT = 3.7	FIG. A. 4
APR					CONT'D
				THE BOEING COMPANY	PAGE 111



CALC		REVISED	DATE	CLMA VS CLA EFFECT OF HORIZONTAL TAIL LAMBDA = 90 DEG. CLT = 3.7	LIMITS EST'D RX-1355 FIG. A.4 CONT'D PAGE 112
CHECK			12/10/76		
APR					
APR					

THE BOEING COMPANY

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CALC		REVISED	DATE	CL vs ALPHA	LE 112 25" H
CHECK			2/10/78	EFFECT OF LANDING GEAR, LAMBDA = 90	RR-13.0
APR				CL = 3.7	FIG. A.5
APR					PAGE 113
				THE BOEING COMPANY	

NOTE
 COMPRESSION CL 5 V.H.
 L3-B01 - 0 DEG.
 INCL ANGLE - 0 DEG.
 THRUST AND ARM FORCES REMOVED

FBI
 100-100000
 100-100000
 100-100000

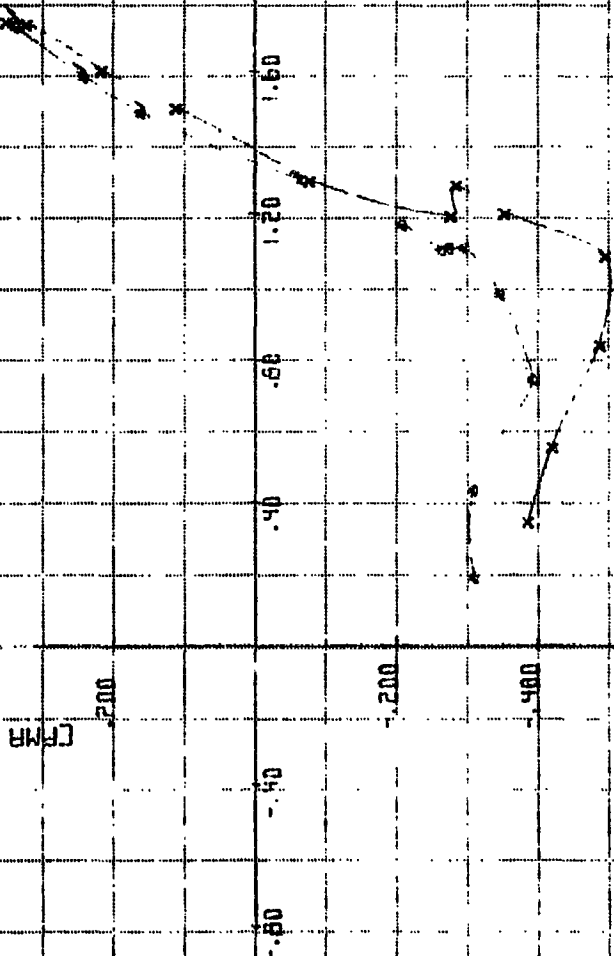
Q
 100-100000
 100-100000
 100-100000

V.M.
 100-100000
 100-100000
 100-100000

Q
 100-100000
 100-100000
 100-100000

Q
 100-100000
 100-100000
 100-100000

Q
 100-100000
 100-100000
 100-100000



CALC	REVISD	DATE
CHECK		07/19/77
APR		
APR		

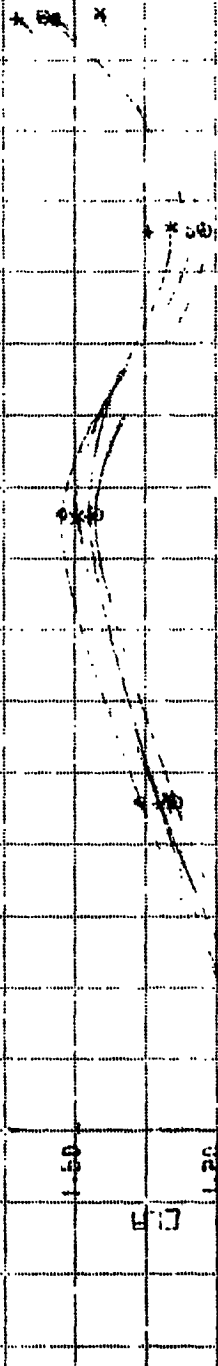
CLM VS CL
 EFFECT OF THRUST LEVEL
 AT CL = CONST.

THE BOEING COMPANY

LSHNB 25-11
 RX-1350
 FIG.A.6
 CONT'D
 PAGE
 118

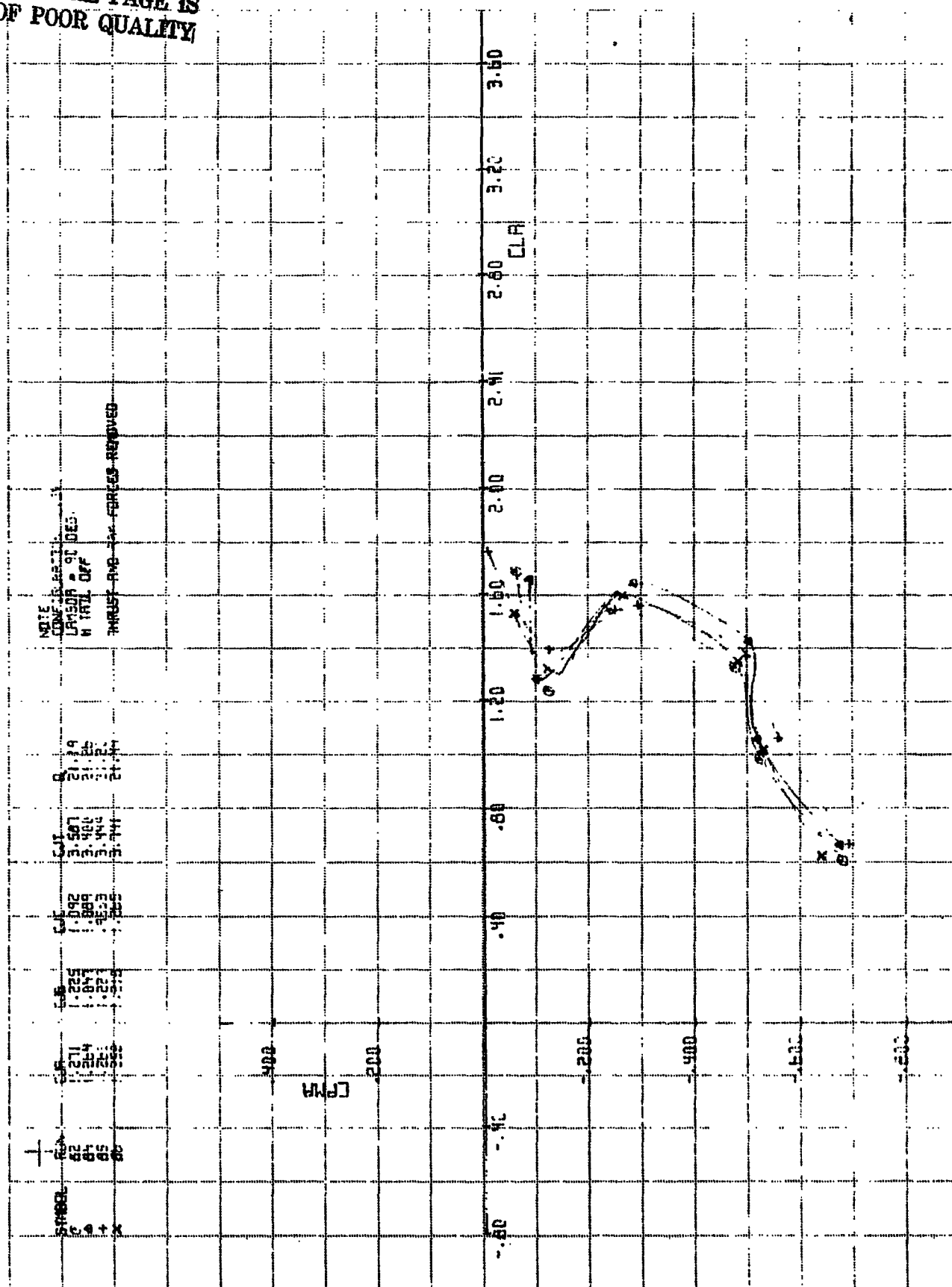
FD
 5.

NAME	DOB	AGE	SEX	HT	WT	HAIR	EYES	SKIN	TEETH	TOES	FEET	HAIR	EYES	SKIN	TEETH	TOES	FEET
ALLEN	11-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
BROWN	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
SMITH	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
JOHNSON	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
WILLIAMS	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
DAVIS	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
MILLER	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
WILSON	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
ANDERSON	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
THOMAS	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
JACKSON	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
WHITE	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
HARRIS	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
CLARK	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
ROBERTS	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
LEE	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
SCOTT	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
GREEN	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
BAKER	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
ADAMS	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
NELSON	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
CARTER	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
MURPHY	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
PERKINS	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
ROBERTSON	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
WATSON	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
WYATT	12-11-19	25	M	5-10	150	B	B	F	1	10	10	B	B	F	1	10	10
COOPER	12-11-19	25	M	5													

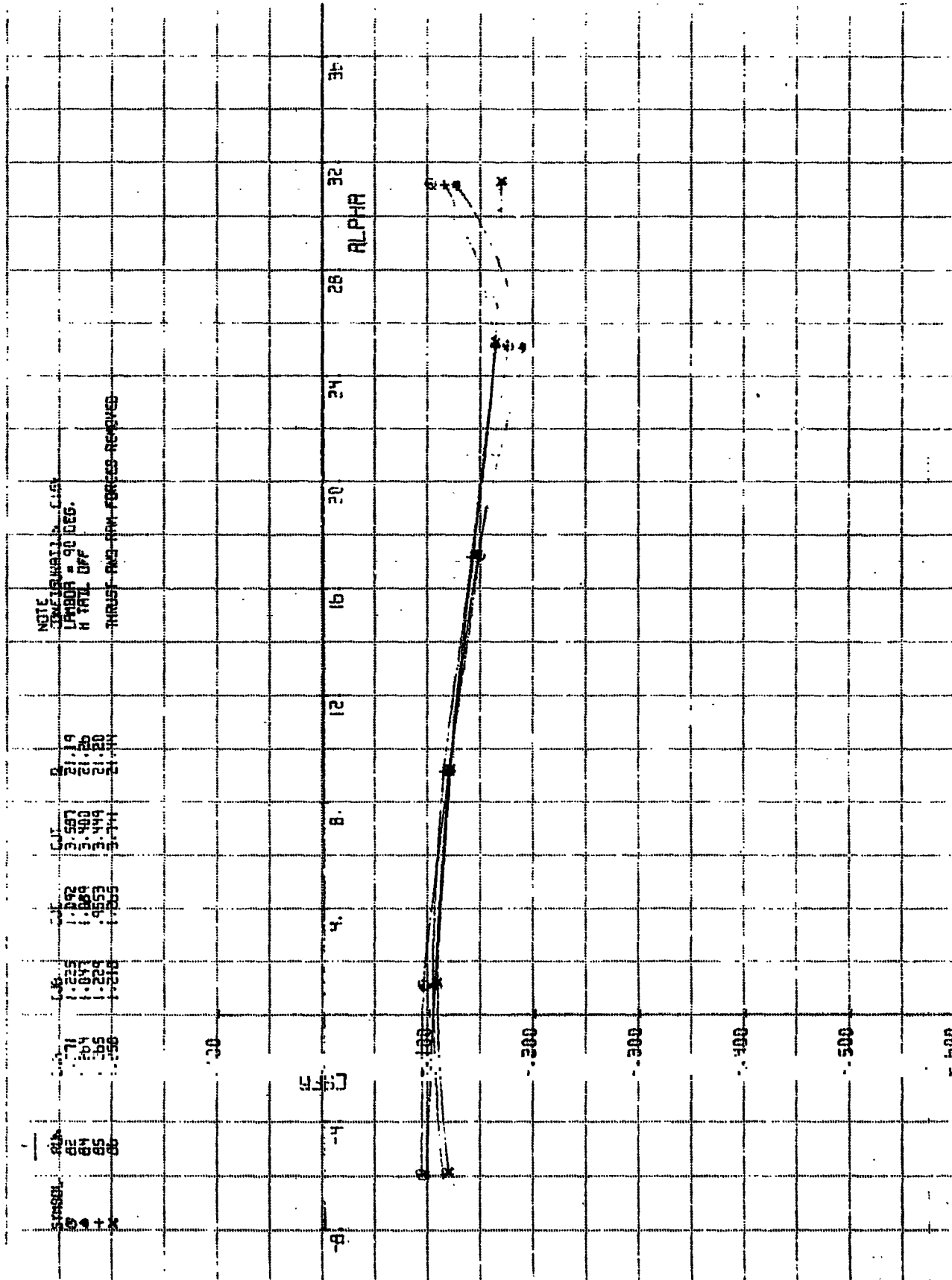


CALC			REVISED	DATE	2.1A VS ALPHA	1010 PSH
CHECK				2/10/71	EFFECT OF DIFFERENTIAL THIN LAMINA = 90 DEG. H. TAIL DEFLECTION	MX-1355
APR						FIG. A. 7
APR					THE BOEING COMPANY	PAGE 119

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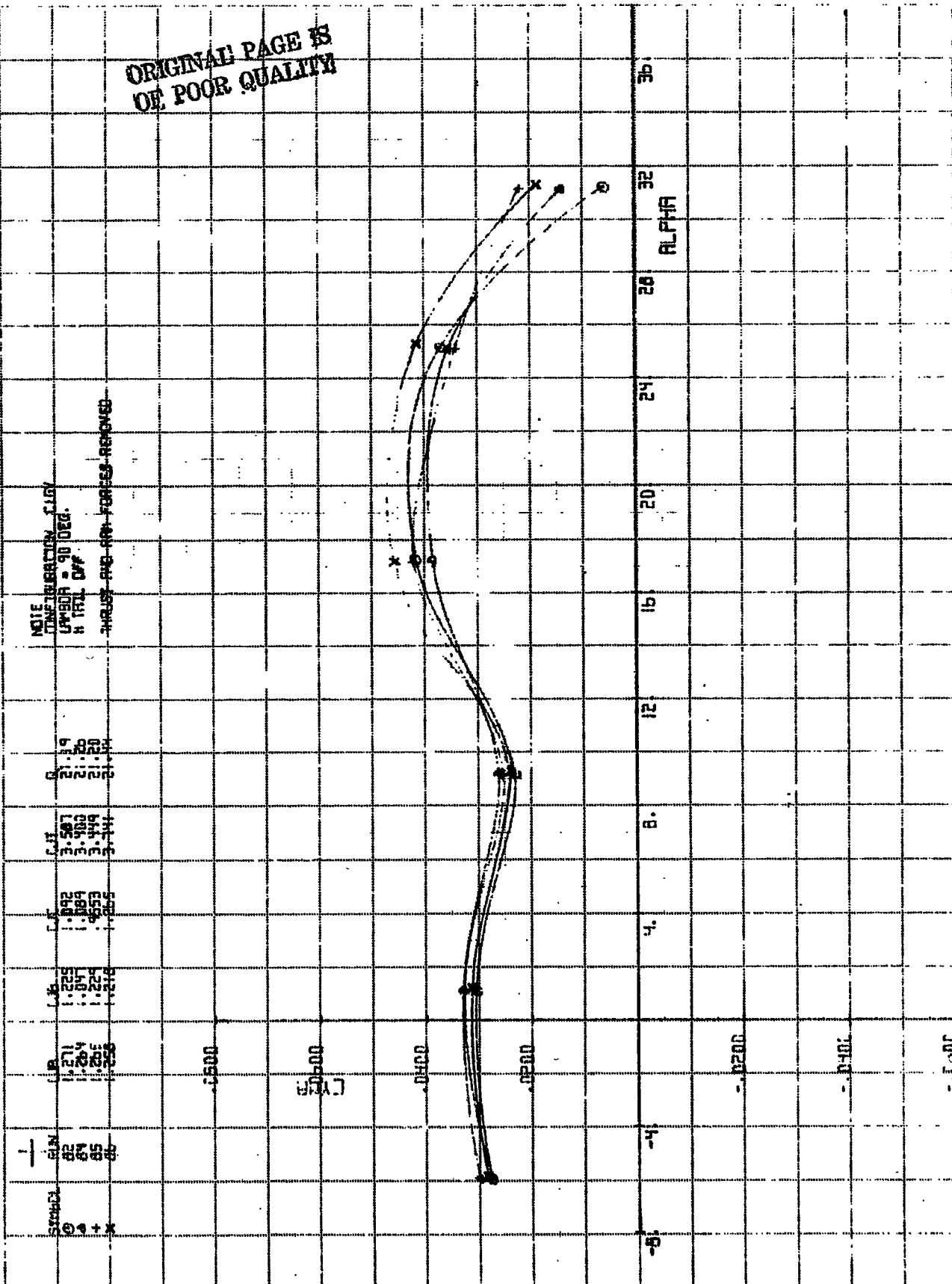


CALC			REVISED	DATE	CMA VS CMA EFFECT OF DIFFERENTIAL THRUST LAMBDA = 90 DEG. H. TAIL OFF.	LSMTB 2544
CHECK				2/15/71		RX-1355
APR						FIG. A.7
APR						CONT'D
					THE BOEING COMPANY	PAGE 121



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NOTE
CONTOURING CLAY
LAMBDA = 90 DEG.
H TAIL OFF
THROAT AND AIR FORCES REMOVED



NOTE:
 1. IN JUNCTION C107
 LAMBO = 90 DEG.
 H TRAIL OFF

THRUST AND AIR FORCES REMOVED

STROD 82 83 84 85
 0.4 + x

CLB 1.211 1.207 1.203 1.200
 1.200 1.200 1.200 1.200

CLF 1.225 1.224 1.223 1.222
 1.222 1.222 1.222 1.222

CLJ 1.237 1.236 1.235 1.234
 1.234 1.234 1.234 1.234

CLK 1.249 1.248 1.247 1.246
 1.246 1.246 1.246 1.246

CLL 1.261 1.260 1.259 1.258
 1.258 1.258 1.258 1.258

CLM 1.273 1.272 1.271 1.270
 1.270 1.270 1.270 1.270

CLN 1.285 1.284 1.283 1.282
 1.282 1.282 1.282 1.282

CLP 1.297 1.296 1.295 1.294
 1.294 1.294 1.294 1.294

CLQ 1.309 1.308 1.307 1.306
 1.306 1.306 1.306 1.306

CLR 1.321 1.320 1.319 1.318
 1.318 1.318 1.318 1.318

CLS 1.333 1.332 1.331 1.330
 1.330 1.330 1.330 1.330

CLT 1.345 1.344 1.343 1.342
 1.342 1.342 1.342 1.342

CLU 1.357 1.356 1.355 1.354
 1.354 1.354 1.354 1.354

CLV 1.369 1.368 1.367 1.366
 1.366 1.366 1.366 1.366

CLW 1.381 1.380 1.379 1.378
 1.378 1.378 1.378 1.378

CLX 1.393 1.392 1.391 1.390
 1.390 1.390 1.390 1.390

CLY 1.405 1.404 1.403 1.402
 1.402 1.402 1.402 1.402

CLZ 1.417 1.416 1.415 1.414
 1.414 1.414 1.414 1.414

CRMA

ALPHA

CL

CL

CL

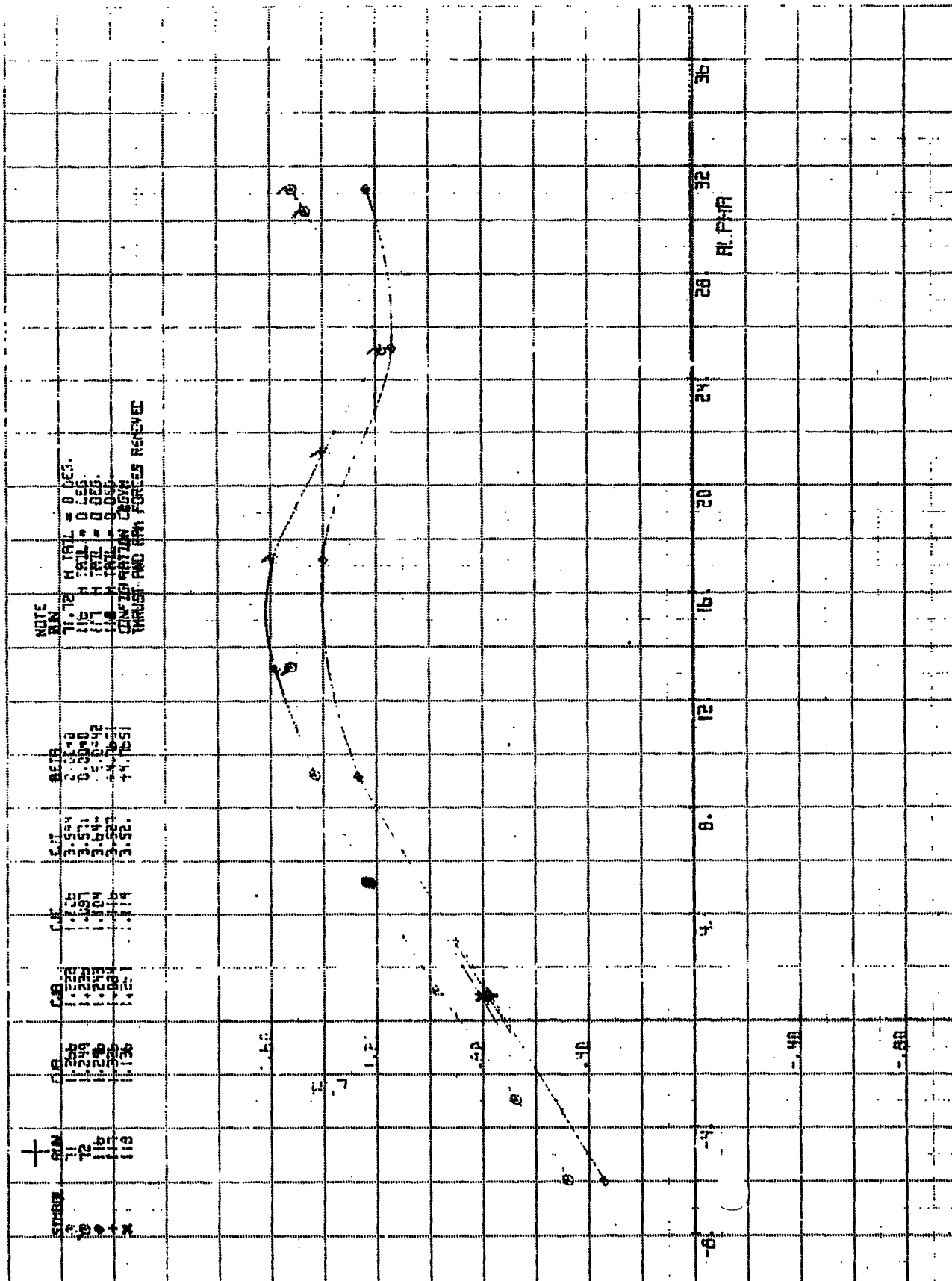
CL

CL

CRMA VS ALPHA
 EFFECT OF DIFFERENTIAL THRUST
 LAMBO = 90 DEG. H. TRAIL OFF

THE BOEING COMPANY

LSMTB 2544
 RX-1553
 FIG. A.7
 CONT'D
 PAGE
 124



NOTE:
 1.12 H. IRL = 0.000
 1.15 H. IRL = 0.000
 1.17 H. IRL = 0.000
 1.18 H. IRL = 0.000
 CONFIRMATION C.O.M.
 THRUST AND RPM FORCES REMOVED

CL

1.6
1.4
1.2
1.0
0.8
0.6
0.4
0.2
0.0

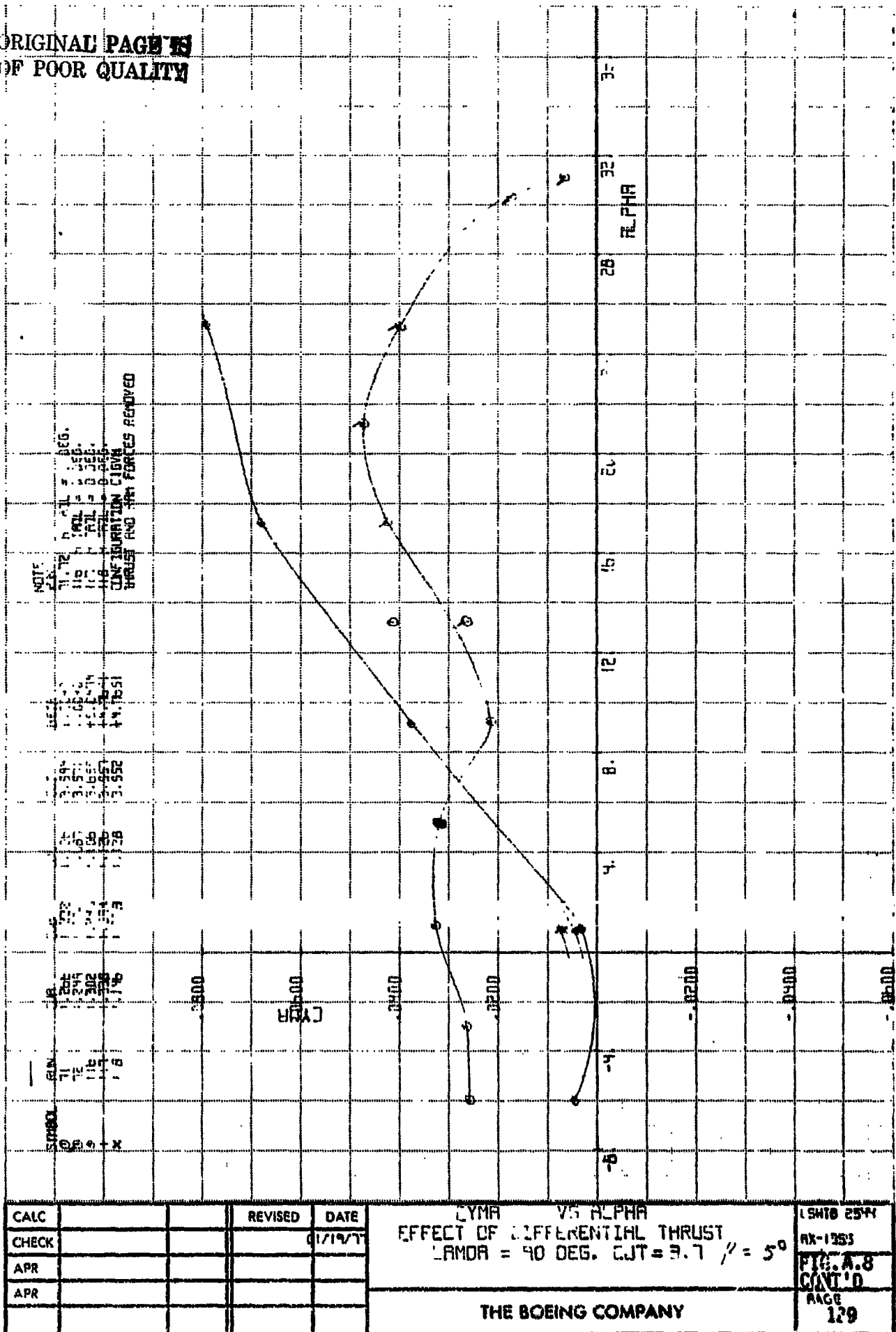
ALPHA

36
32
28
24
20
16
12
8
4
0

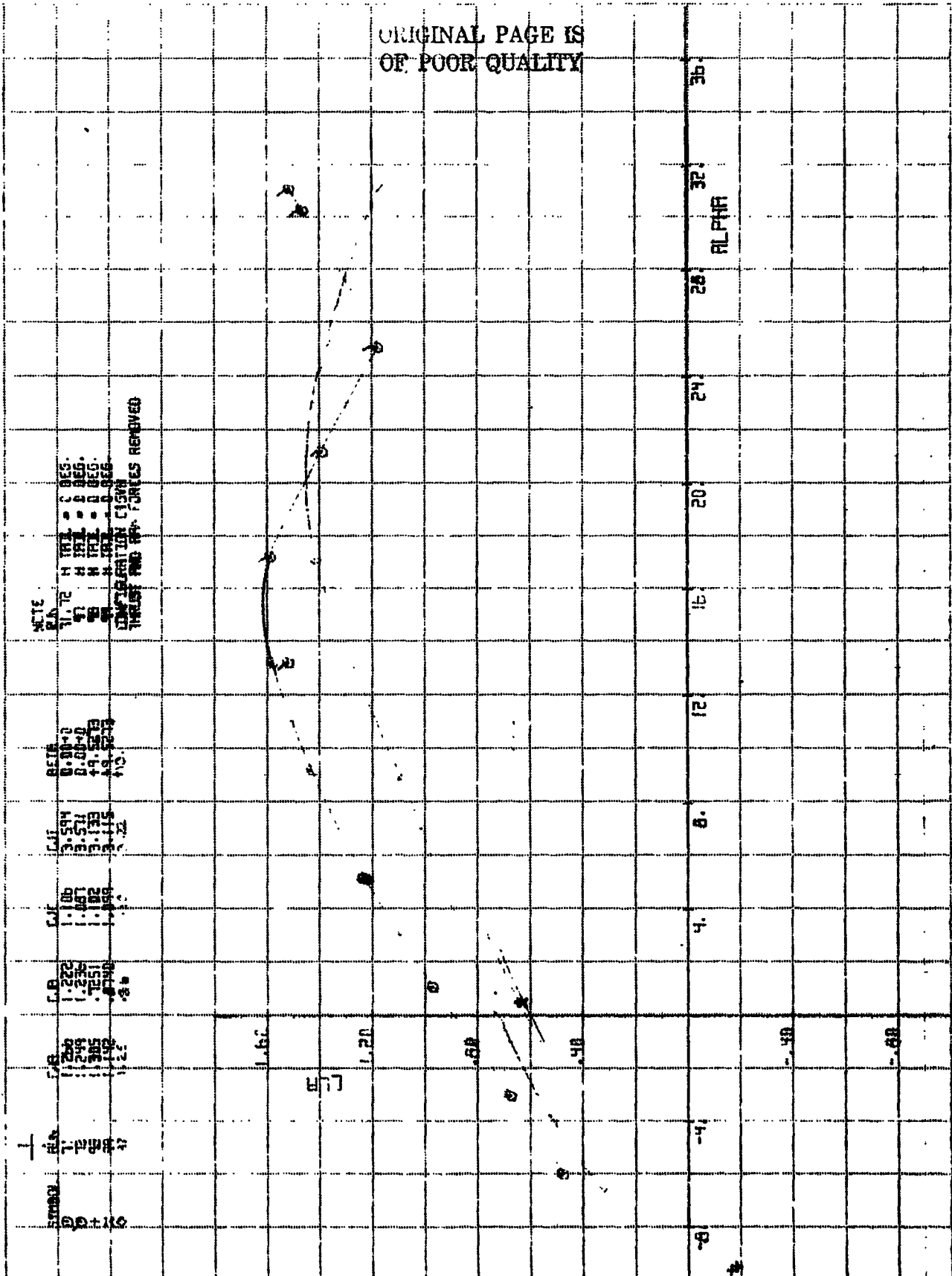
CALC		REVISED	DATE	CL vs ALPHA EFFECT OF LIFT COEFFICIENT THRUST LAMBDA = 1.0, C.T. = 4.1, $\beta = 5$	LEWIS 274
CHECK			2/10/76		PM-1355
APR					FIG. A. 8
APR					PAGE 125

THE BOEING COMPANY

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CALC		REVISED	DATE	CL VS ALPHA	LSM 2744
CHECK			12/10/78	EFFECT OF DIFFERENTIAL THRUST	RA-1352
APR				LAMDA = 90 DEG, CDT = 3, $\beta = 10$	FIG.A.9
APR					PAGE 131
THE BOEING COMPANY					

NOTI
IN
71
91
88

H TEL = 0-655.
H TEL = 0-655.
H TEL = 0-655.

CONFIRMATION FROM
THRUST AND REM FORCES ADVISED

11-25-64
11-25-64
11-25-64
11-25-64

3. ၁၆၆၆

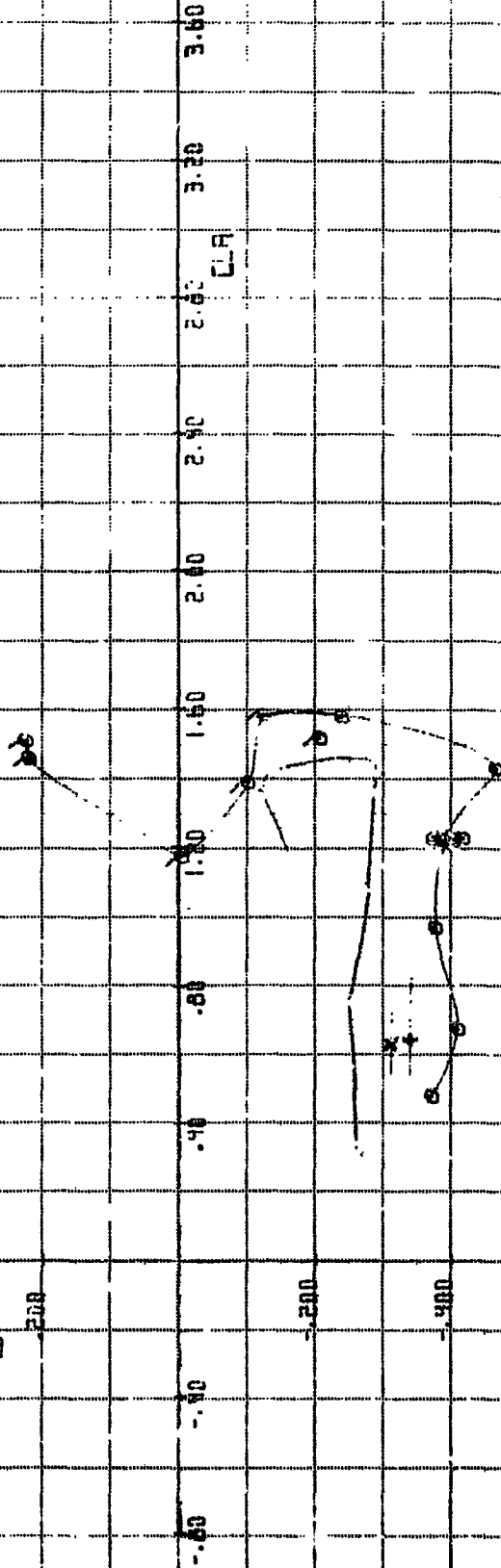
ॐ नमो भगवते वासुदेवाय

522.1
522.1
522.1

புதுச்சேரி
புதுச்சேரி நகராட்சி
புதுச்சேரி நகராட்சி
புதுச்சேரி நகராட்சி

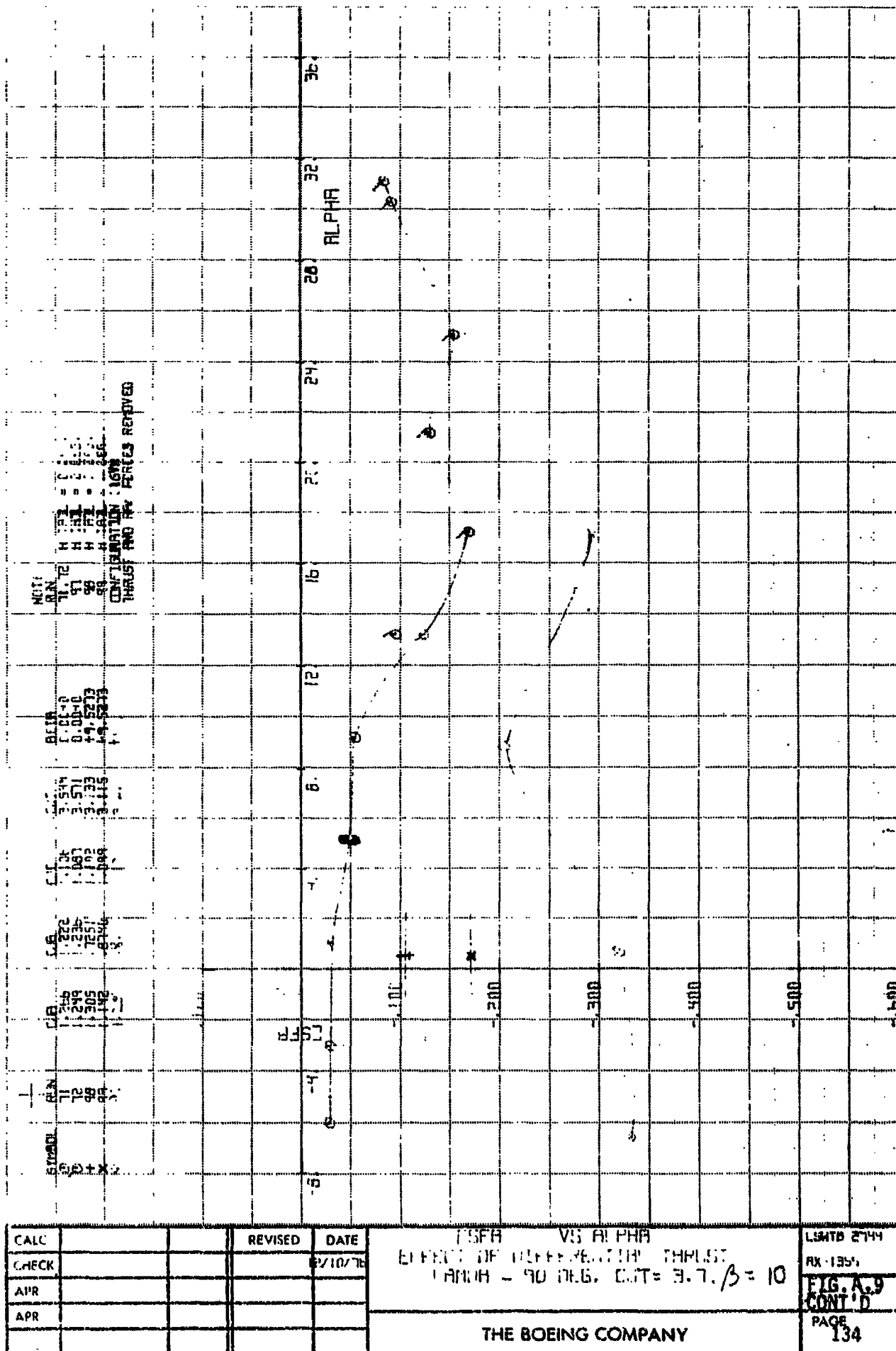
十、

45129

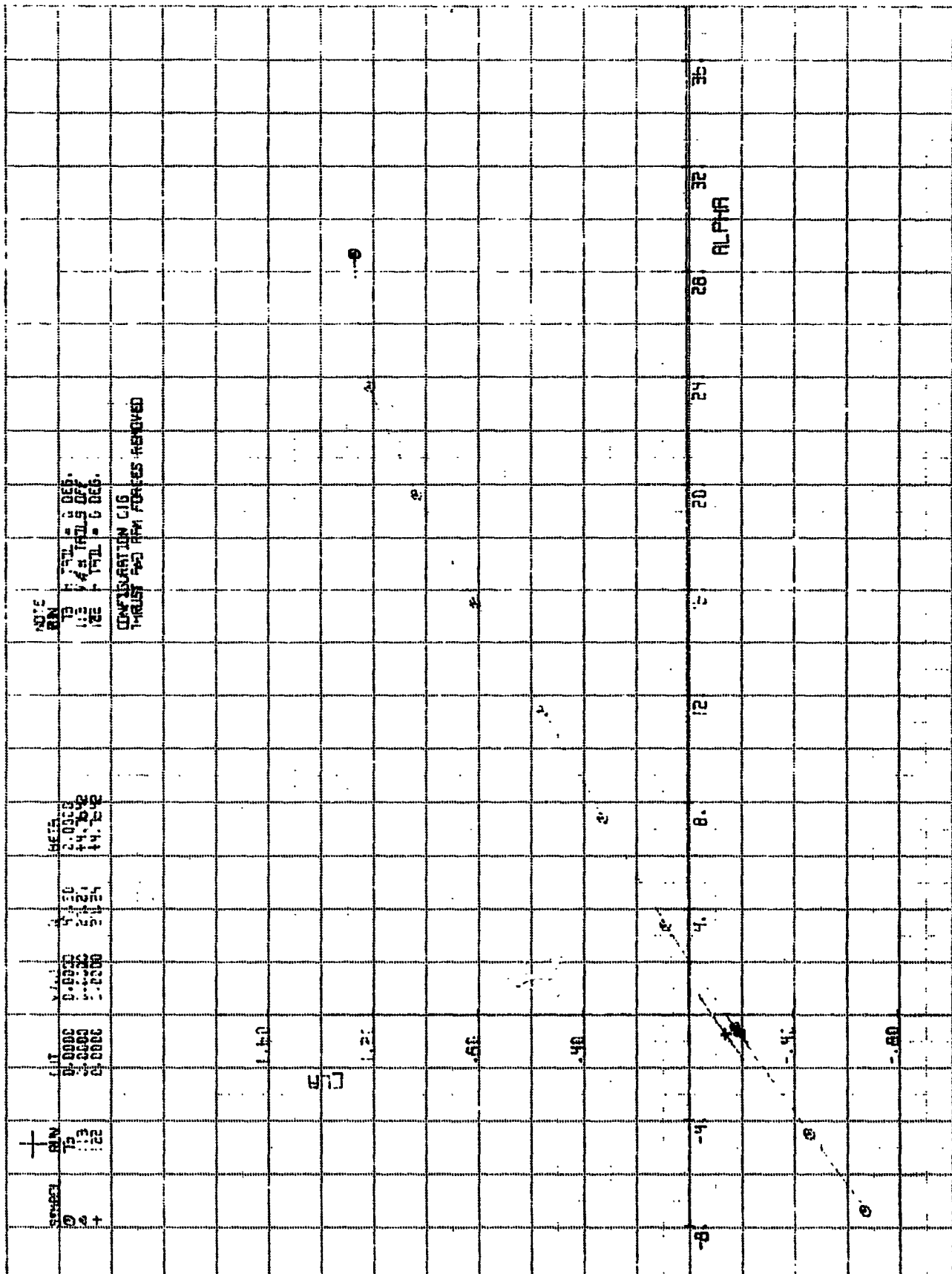


CALC			REVISED	DATE	CPMA VS CLA	LSMTD 2TH
CHECK				2/10	EFFECT OF DIFFERENTIAL THRUST	AS-1335
APR					LAMDA = 90 DEG. CJT = 3.1, $\beta = 10$	FIG. A-9
APR						CONT'D
					THE BOEING COMPANY	PAGE 133

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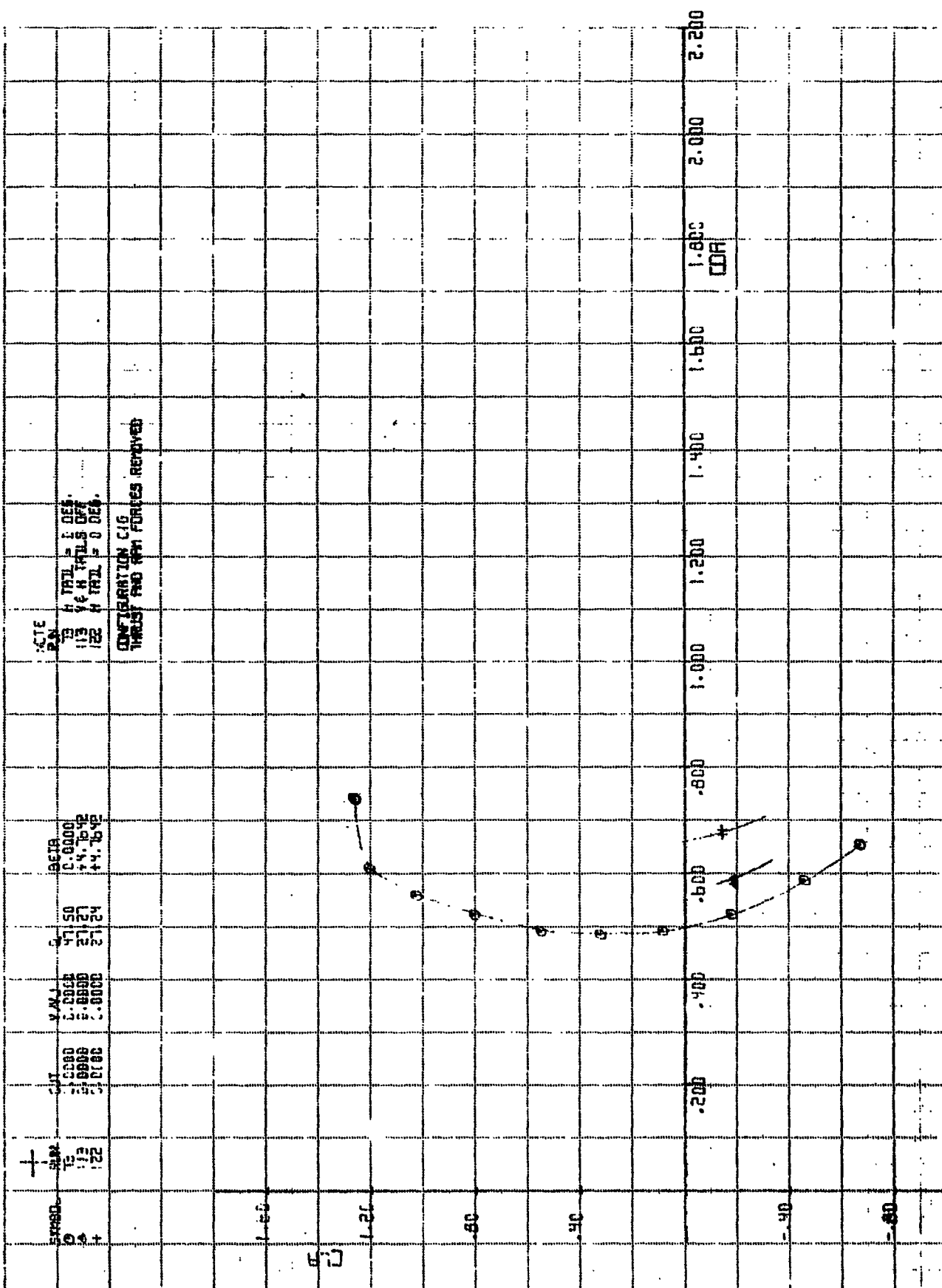
CALC		REVISED	DATE	1058 V-10 ALPHA	LSHED 2744
CHECK			8/10/76	EFFECT OF DIFFERENTIAL THRUST	RX-135
APR				ALPHA = 90 DEG. ENT = 3.7. $\beta = 10$	FIG. A.9
APR					CONT'D
				THE BOEING COMPANY	PAGE 134



CLM	VS ALPHA	LSMTB 2544
VERTICAL TAIL EFFECTIVENESS		RX-1355
LAMDA = 90 DEG, CJT = 0		FIG.A.10
THE BOEING COMPANY		PAGE 137

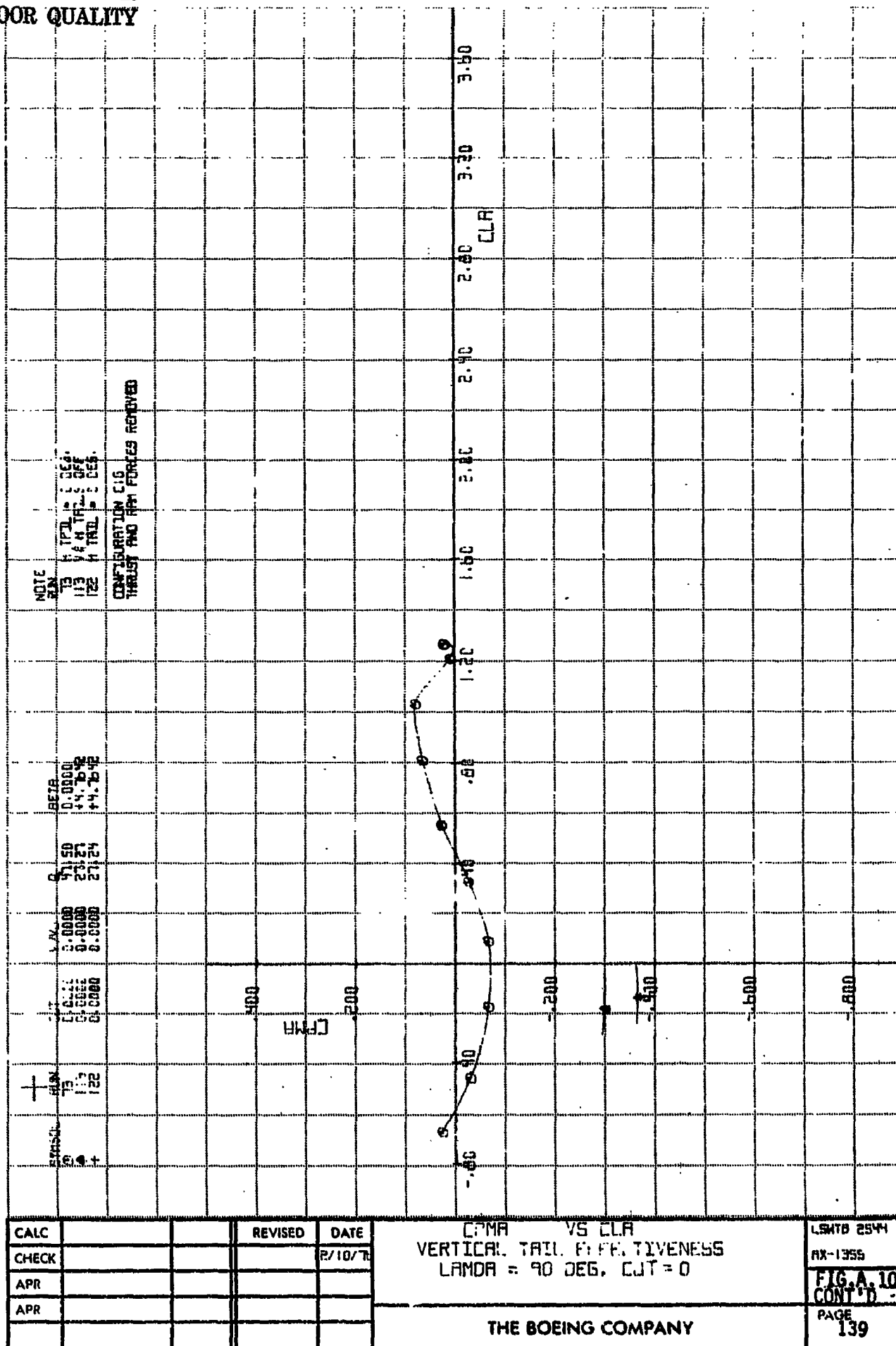
CALC		REVISED	DATE
CHECK			2/10/76
APR			
APR			

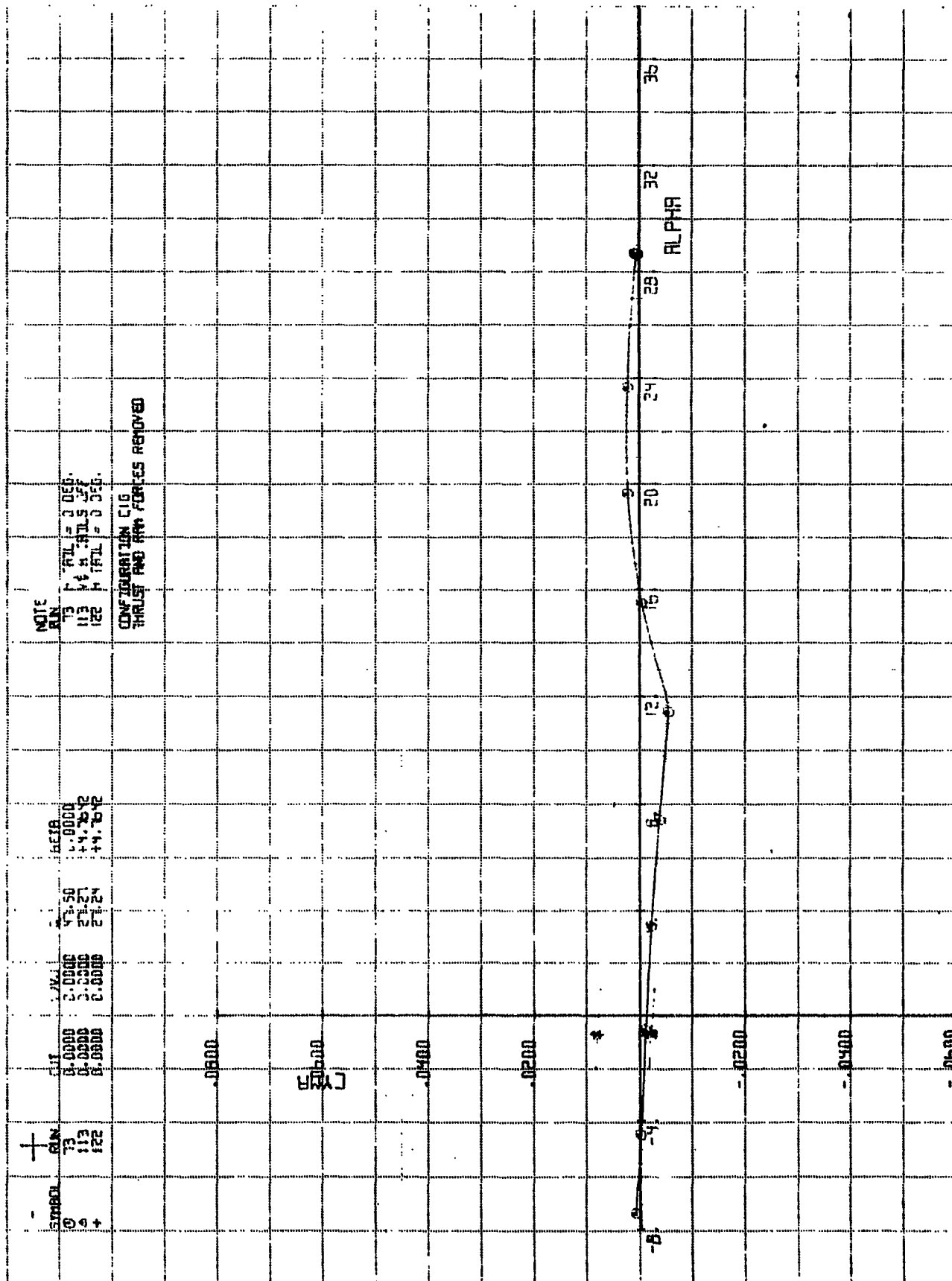
FIG 37
37.



CALC		REVISED	DATE	CL VS CDR VERTICAL TAIL EFFECTIVENESS LAMBDA = 90 DEG, CLT = 0	LSHTB 2544
CHECK			2/10/78		RX-1355
APR					FIG. A-10
APR					CONT'D
				THE BOEING COMPANY	PAGE 138

FILE: 7:
40.





CALC		REVISED	DATE	LYMR VS ALPHA	LSMTB 25M
CHECK			01/14/71	VERTICAL TAIL EFFECTIVENESS	RX-1375
APR				ARMCH = 40 DEG, CJT = 0	FIG. A.10
APR					CONT'D
				THE BOEING COMPANY	PAGE 141

MOT: H. TAIL ANGLE
 RUN: 0.000
 90 VEH TAIL FF
 102 0 DEG. CL 0
 LONGITUDINAL CL 0
 THRUST AND RPH FORCES REMOVED

ACIR
 0.0000
 +1.3234
 +1.5234

V X V
 0.0000
 0.0000
 0.0000

C U
 0.0000
 0.0000
 0.0000

RUN
 73
 90
 102

STRUT
 0
 0
 0

1.60

1.20

.80

.40

0

-.40

CL

2.200
2.000
1.800
1.600
1.400
1.200
1.000
.800
.600
.400
.200
0
-.200
-.400
-.600
-.800

COA

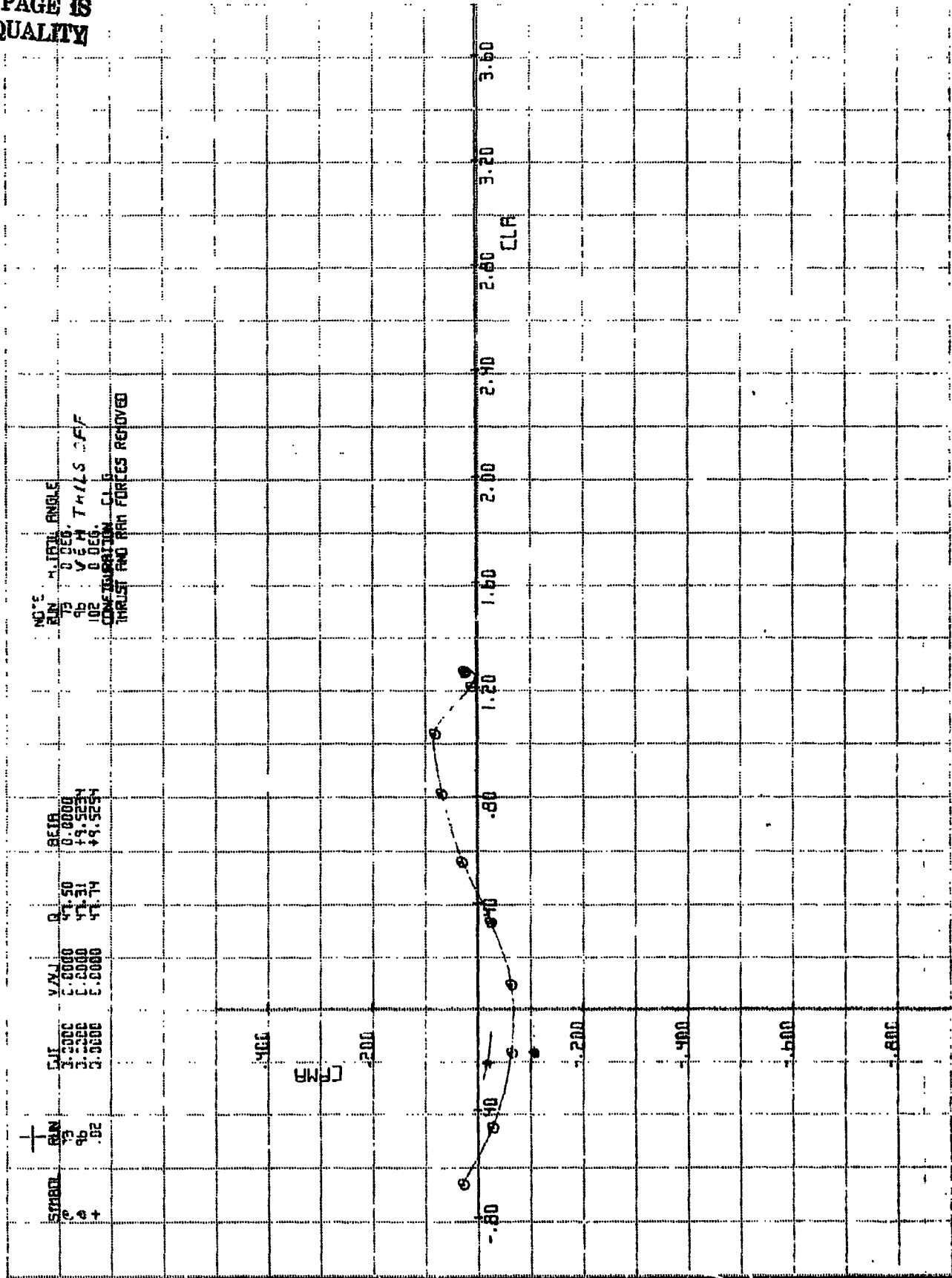
CALC		REVISED	DATE
CHECK			
APR			
APR			

FLR VS COA
 VERTICAL TAIL EFFECTIVENESS
 HMBOR = 90 DEG. CL = 0

LSMTB 25-44
 AX-1355
 FIG. A. 11
 CONT'D
 PAGE
 144

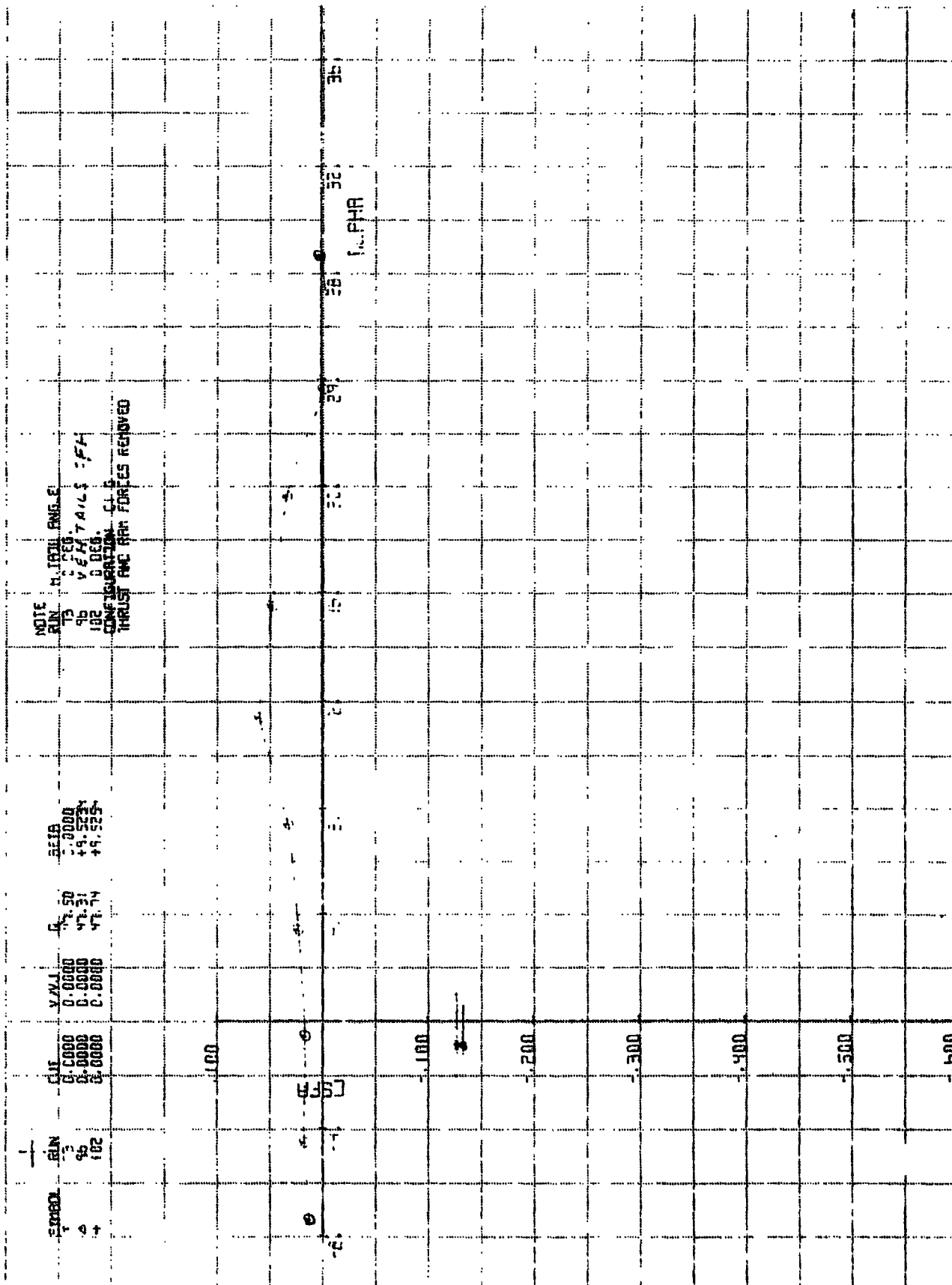
THE BOEING COMPANY

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CALC		REVISED	DATE	CPMA VS CLa VERTICAL TAIL EFFECTIVENESS LAMBDA = 90 DEG. CL = 0	LSM10 25-4
CHECK			01/15/77		AX-1355
APR					FIG. A.11
APR					CONT'D
				THE BOEING COMPANY	PAGE 145

FIG 38



CALC			REVISED	DATE	VS ALPHA	5418 P544
CHECK				3/15/77	VERTICAL THRUST EFFECTIVENESS	HX-1355
APR					LAMBDA = 40 DEG.	FIG. A.11
APR					CJ = 0	CONT'D
					THE BOEING COMPANY	PAGE 146

NOTE: H. 1801. 8016.
 13 0 DEG.
 96 1/2 DEG.
 102 0 DEG.
 CONTOUR LINE CL-18
 THRUST AND AIR FORCES REMOVED

SEIA
 0.0000
 +9.5234
 +9.5234

Q
 47.50
 47.31
 47.14

V/V
 0.0000
 0.0000
 0.0000

CJ
 0.0000
 0.0000
 0.0000

STAB
 13
 96
 102

36

32

28

24

20

16

12

8

4

0

-4

-8

-12

-16

-20

ALPHA

CALC		REVISED	DATE
CHECK			01/15/77
APR			
APR			

CRMA VS ALPHA
 VERTICAL TAIL EFFECTIVENESS
 LAMBDA = 90 DEG. CJ = 0

LSHTB 25-M
 AX-1355
 FIG. A.11
 CONT'D
 PAGE 148

THE BOEING COMPANY

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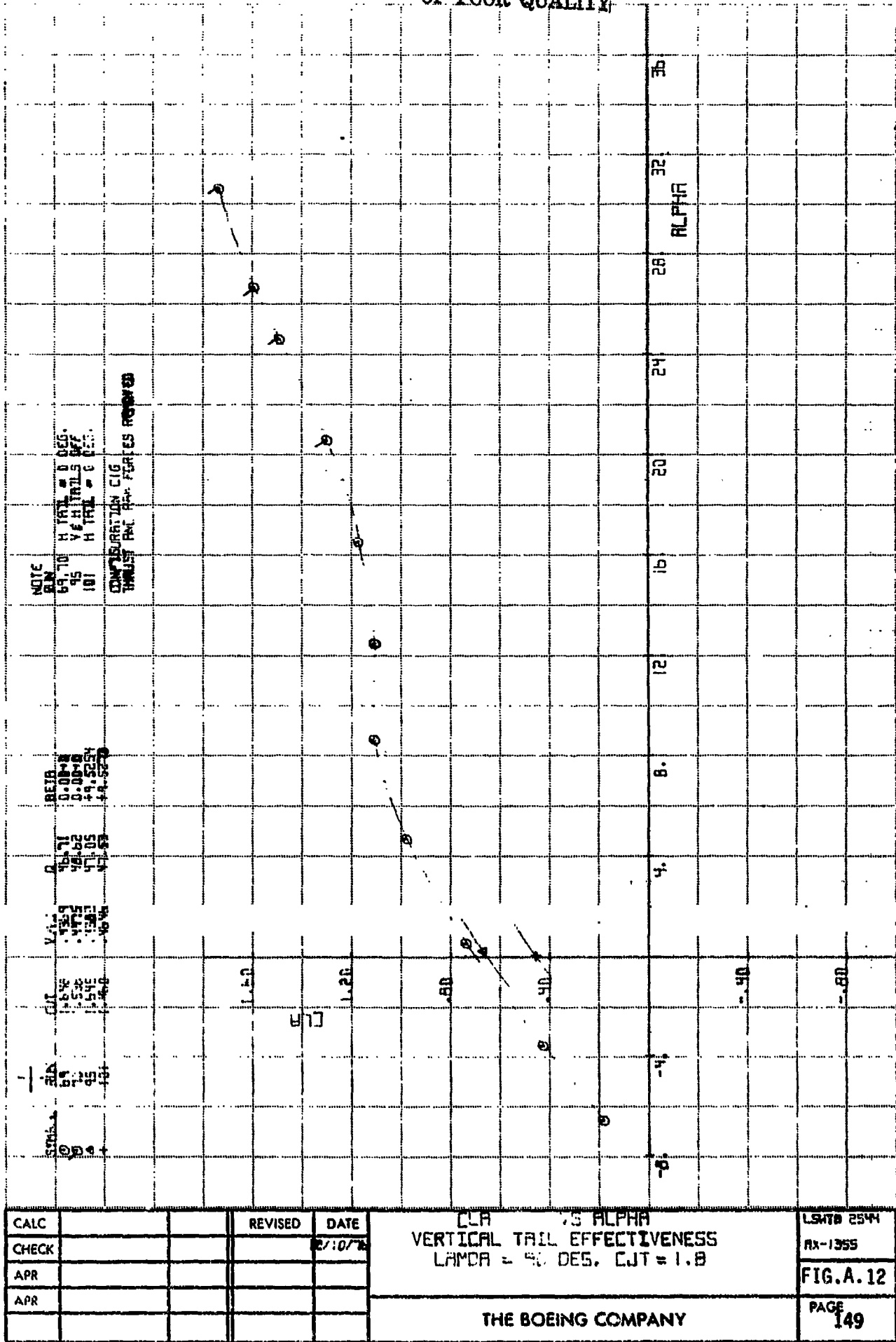
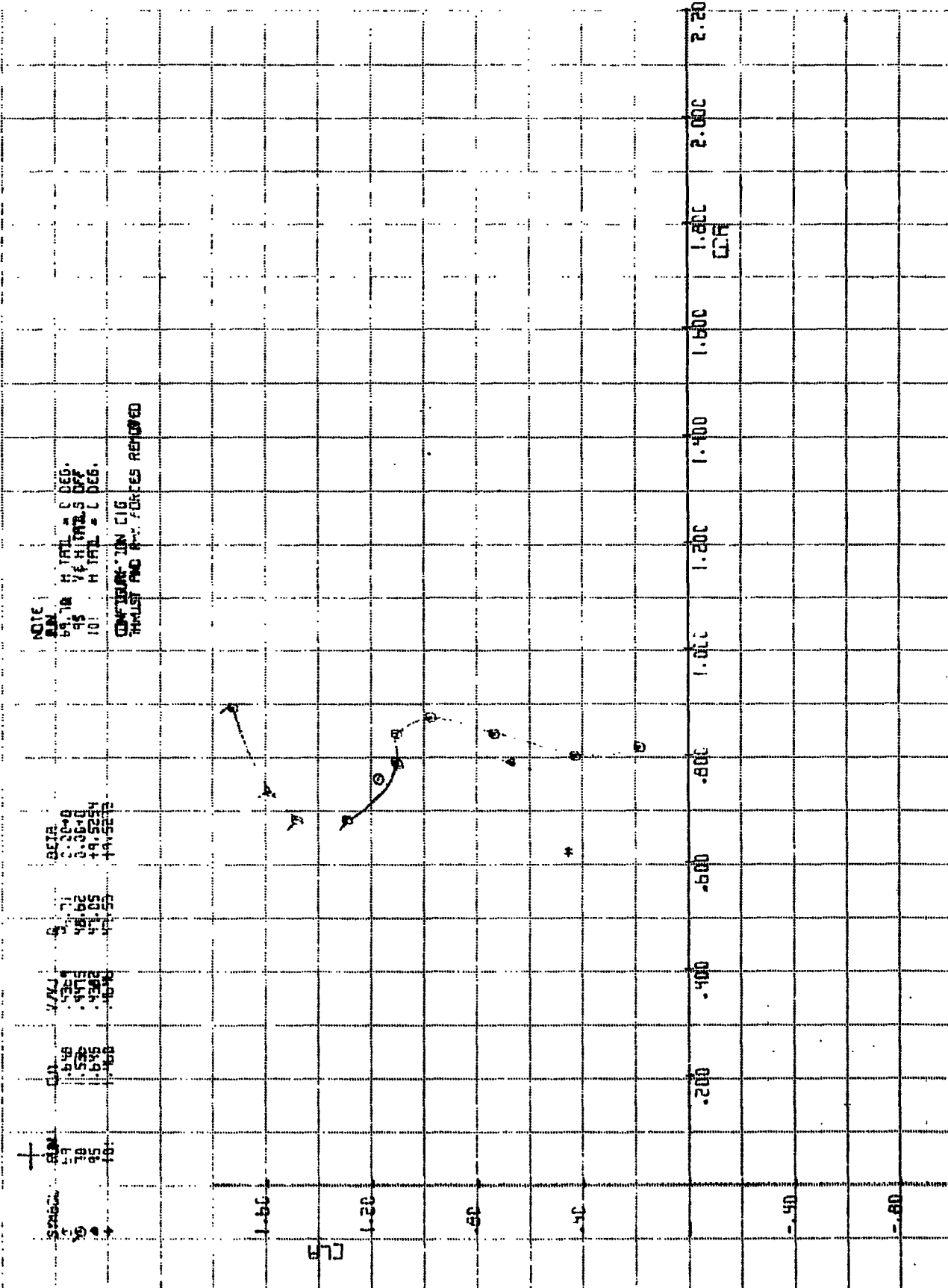


FIG 35
25.



CALC	REVISD	DATE	LLA VS LNA VERTICAL TAIL EFFECTIVENESS LAMDA = 70 DEG. C/T = 1.8	LSHTB 2544
CHECK		2/10/78		RS-1355
APR				FIG. A 12
APR				PAGE 150

THE BOEING COMPANY

01.69.70
56
101
H TATL - 1 DES.
V H TATL S OFF
H TATL - 1 DEG.

CONFIDENTIAL: CIG
THREAT AND ARM FORCES REMOVED

44-38861-1000

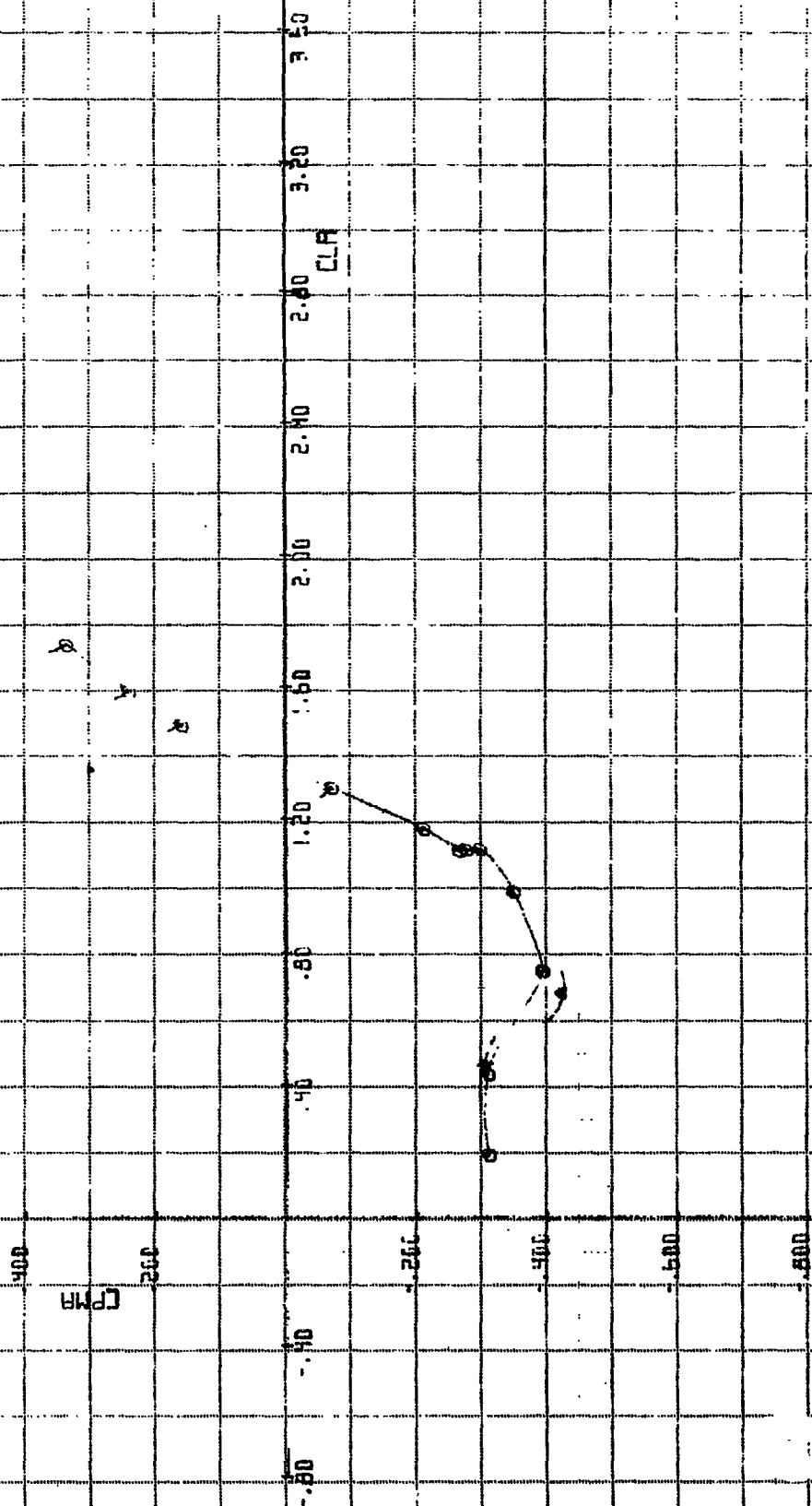
46.71
48.62
47.05
47.52

2025-03-27

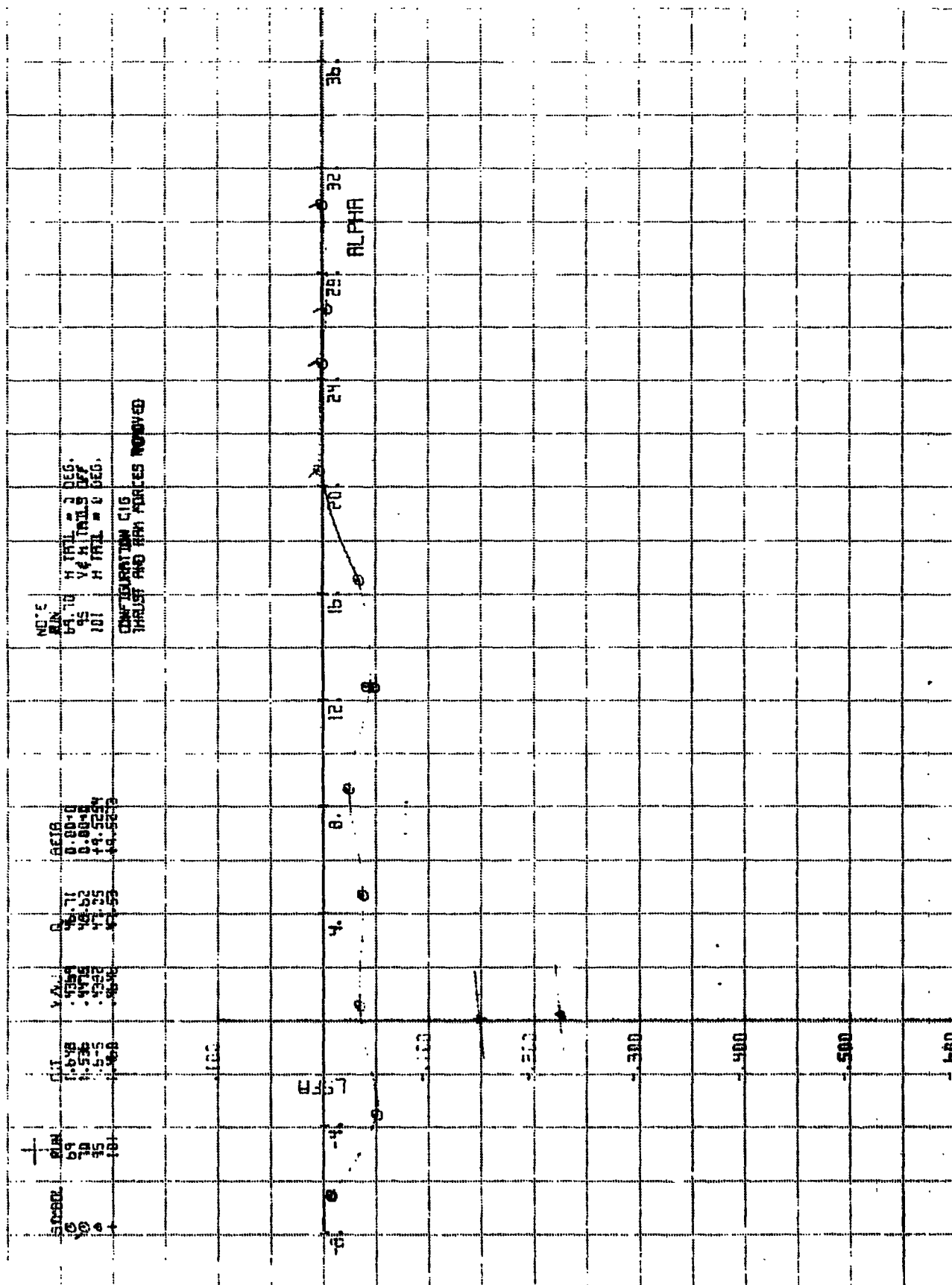
1-48 5-48 6-48

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7080

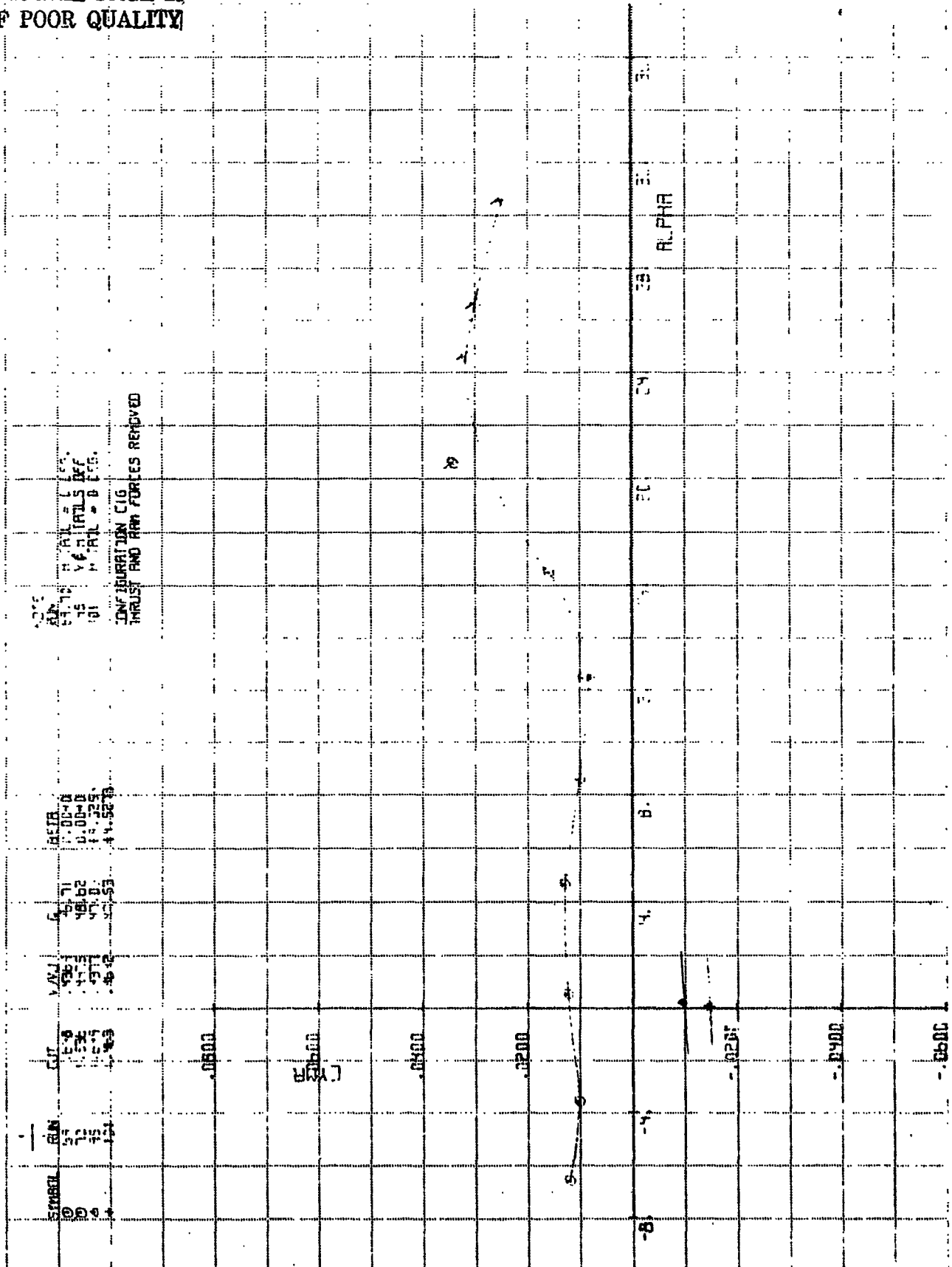


CALC			REVISED	DATE	CPMA VS CLH VERTICAL TAIL EFFECTIVENESS LAMDA = 90 DEG, CUJ = 1.8	LSMTB 2544
CHECK				2/10/78		PN-1355
APR						FIG. A. 12
APR						CONT'D
					THE BOEING COMPANY	PAGE 151



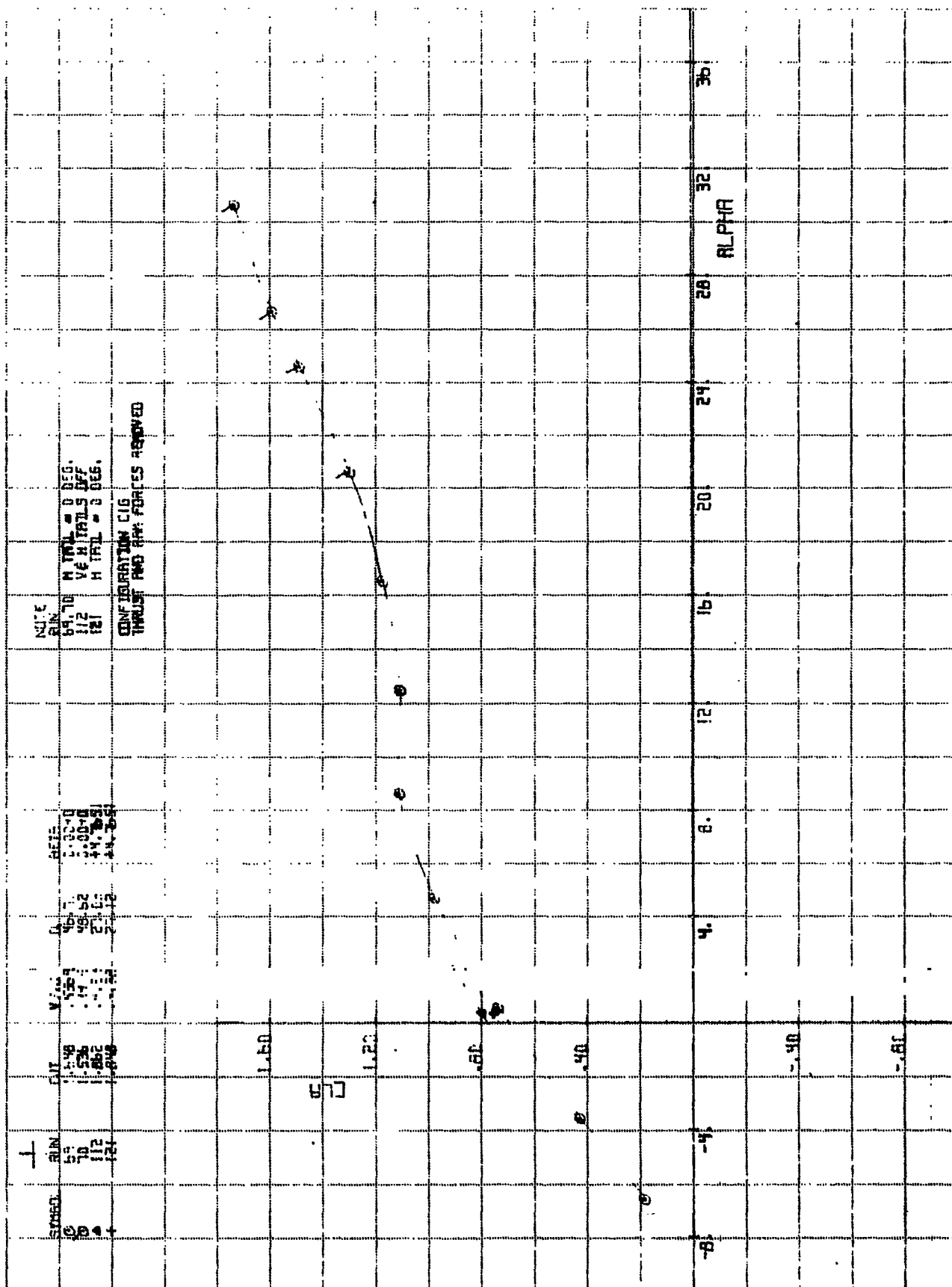
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CALC																			
CHECK																			
APR																			
APR																			
REVISED	DATE																		

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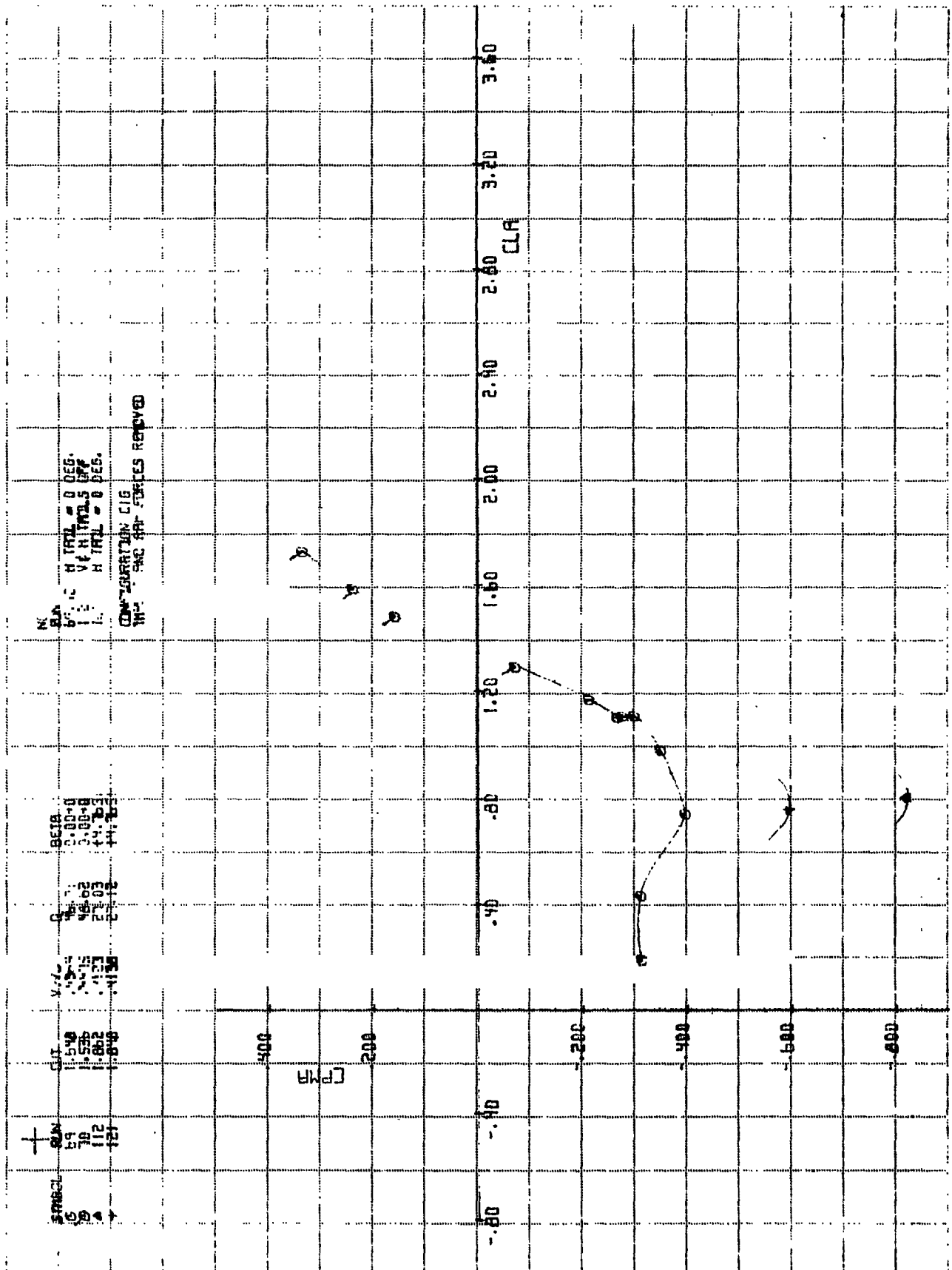


CALC		REVISED	DATE	CYPR VS ALPHA VERTICAL TAIL EFFECTIVE ARMOR - 40 DLS. C/T = 1.8	LSMTB 25-11
CHECK			1/19/77		RX-1355
APR					FIG.A.12
APR					CONT'D
				THE BOEING COMPANY	PAGE 153

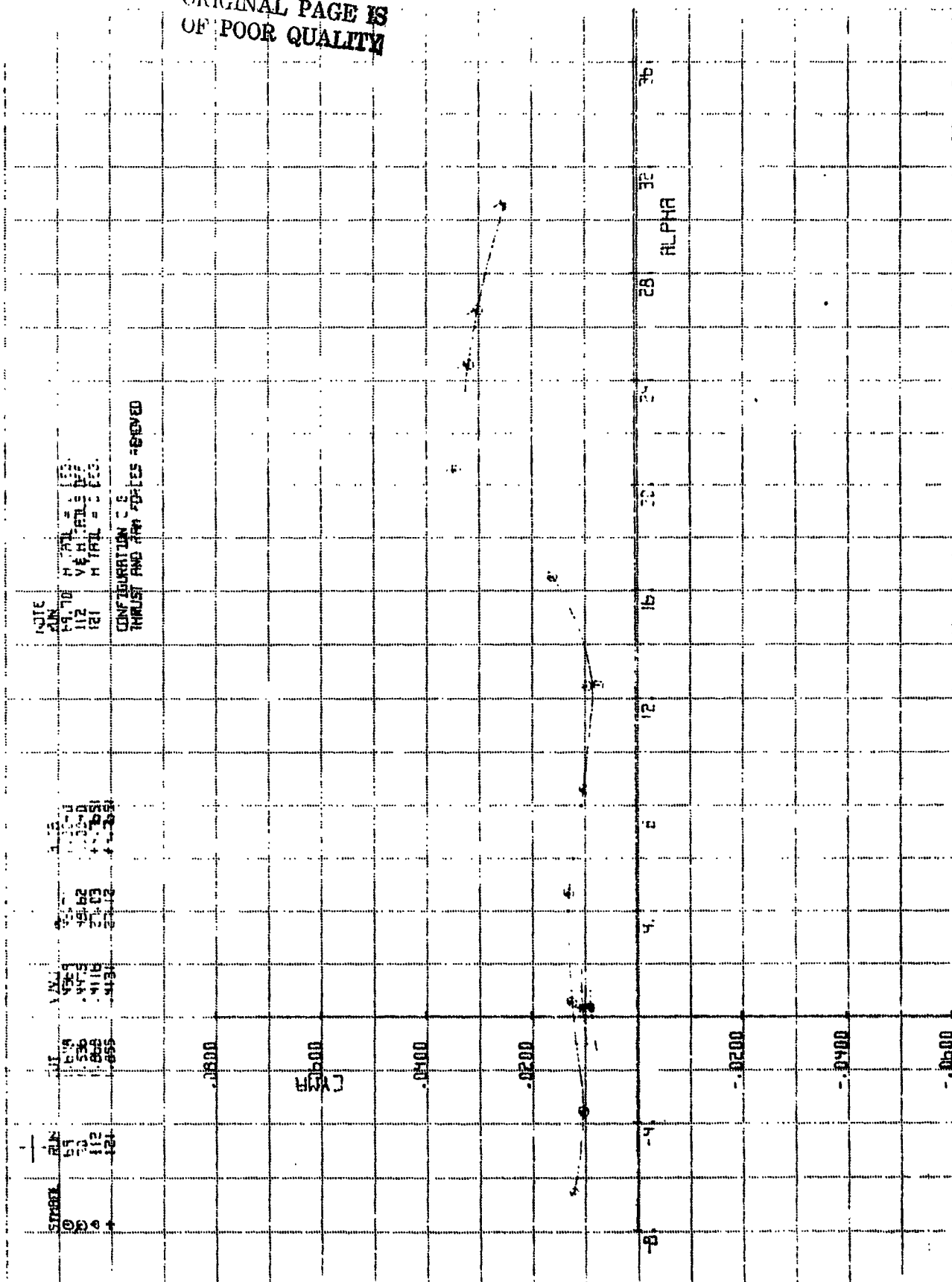
FIG 3.3
5.



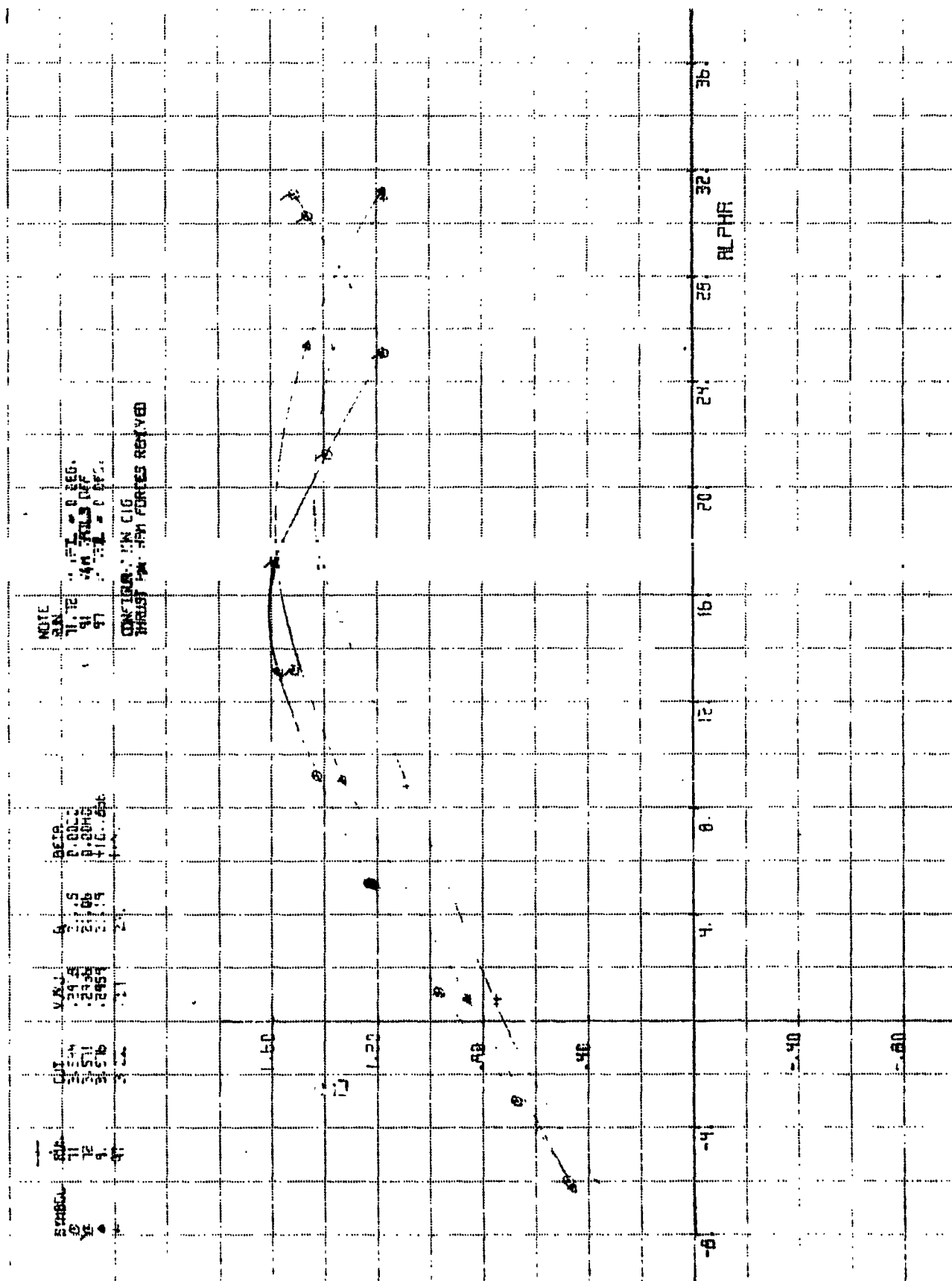
CLC		REVISED	DATE	CLH VS ALPHA	LEWIS 2544
CHECK			2/10/72	VERTICAL TAIL EFFECTIVENESS	RX 1355
APR				LAMDA = 90, C.T. = 1.8	FIG. A.13
APR				THE BOEING COMPANY	PAGE 155



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CALC		REVISED	DATE	LYMA V. ALPHA	LSW18 2544
CHECK			01/19/77	VERTICAL TAIL EFFECTS	AX-1335
APR				IMDR = 90 DEG. LUT = 1.8	FIG. A.13
APR					CONT'D
				THE BOEING COMPANY	PAGE 159

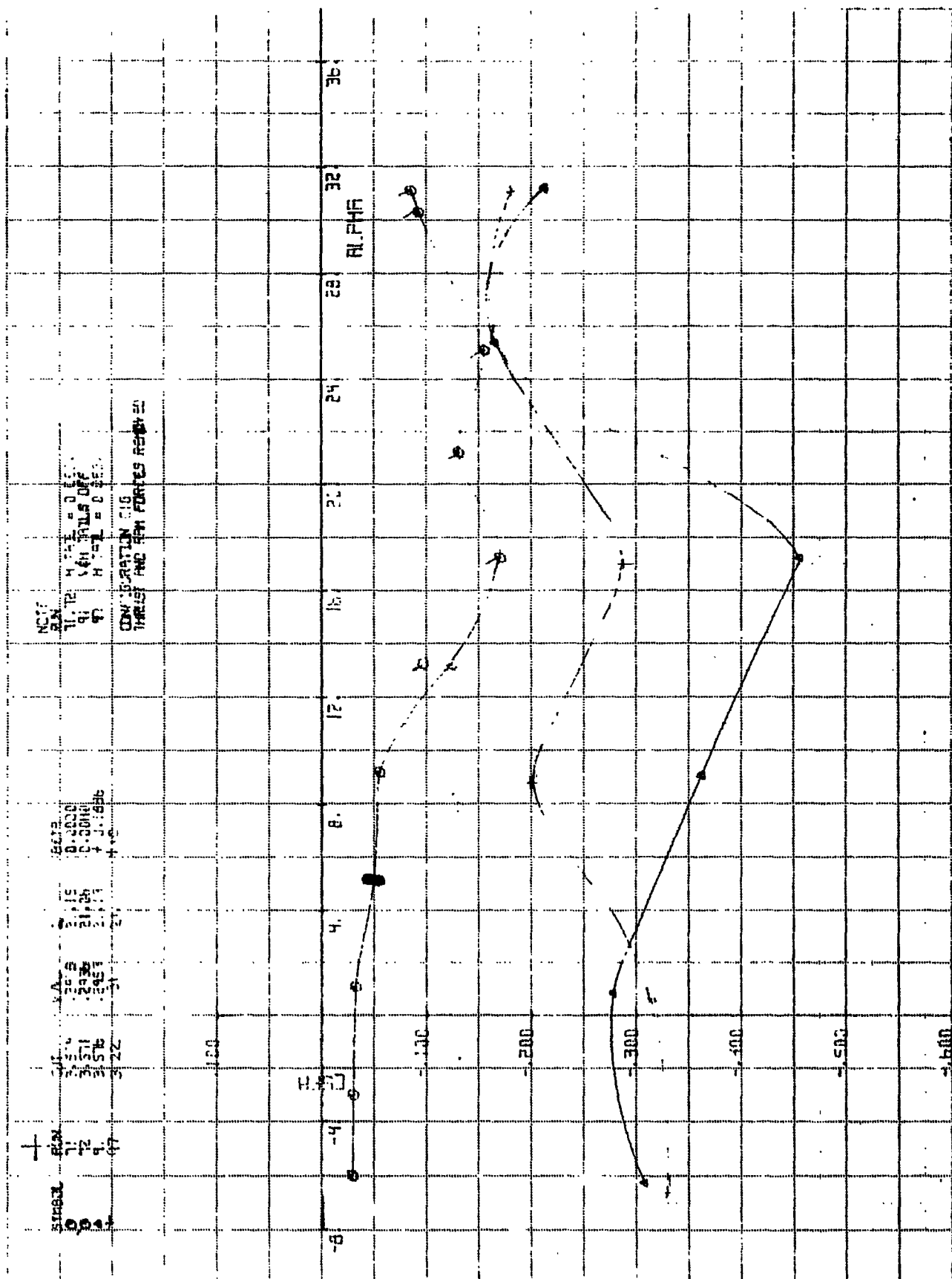


CALC		REVISED	DATE	CL vs ALPHA	LIMIT 2544
CHECK			2/10/76	VERTICAL TAIL EFFECTIVENESS	HX-1953
APR				LAMDA = 90 DEG. C/T = 3.7	FIG. A.14
APR					PAGE 161
THE BOEING COMPANY					

TIME	ALT	WIND	WAVE	DATA	NOTE
0	3159	237	2115	11.72	H. FAL - 0.05G.
1	3151	233	2116	91	VAN TAILS OFF.
2	3157	245	2119	91	H. FAL - 0.05G.
3	3122	21	21		CONFIGURATION CIG
					THRUST AND RPM FORCES ADJUSTED

CONFIGURATION CIG
THRUST AND RPM FORCES ADJUSTED

CALC			REVISED	DATE	LAMDA VS CLRA VERTICAL TAIL EFFECTIVENESS LAMDA = 90 DEG, CJT = 3.1	LSMTD 2544
CHECK				2/10/76		RX-1955
APR						FIG. A-14
APR						CONT. 41
					THE BOEING COMPANY	PAGE 163



CAIC		REVISED	DATE	FSFH VS ALPHA VERTICAL TAIL EFFECTIVENESS LAMBDA = .0046, CDT = 3.1	LEH 2544 RX-1955
CHECK			2/10/76		FIG. A.14 CONT'D
APR					PAGE
APR				THE BOEING COMPANY	164

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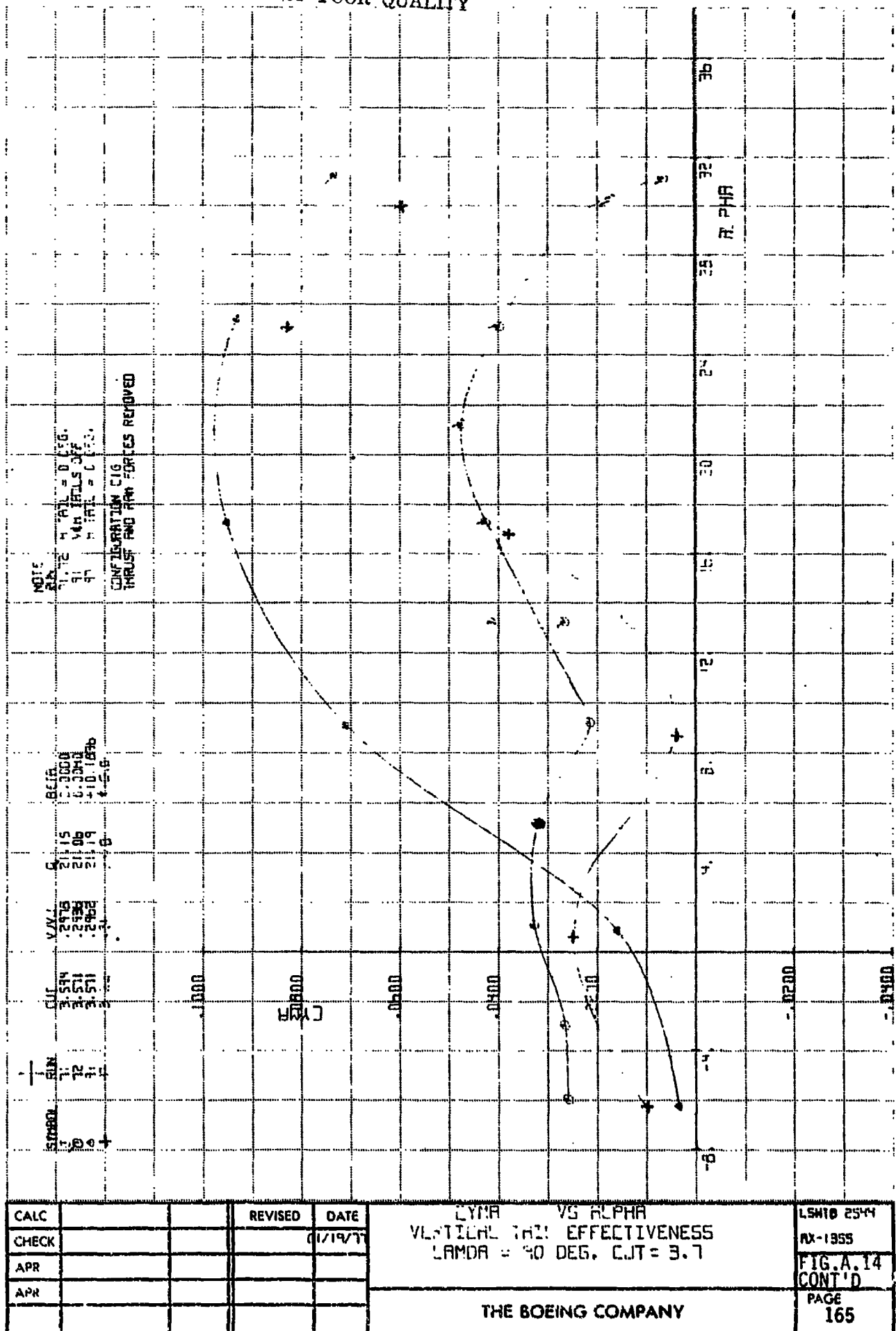


FIG 31
1.

CALC		REVISED	DATE	CYMA VS ALPHA VERTICAL TAIL EFFECTIVENESS LAMDA = 90 DEG. C/T = 3.7	LSMTB 2544
CHECK			01/19/77		AX-1355
APR					FIG. A.14
APR					CONT'D
				THE BOEING COMPANY	PAGE 165

CALC			REVISED	DATE
CHECK				2/10/7
APR				
APR				

CRMA VS ALPHA
VERTICAL TAIL EFFECTIVE WAVE
LENGTH = 90 DEG, CUT = 3.7

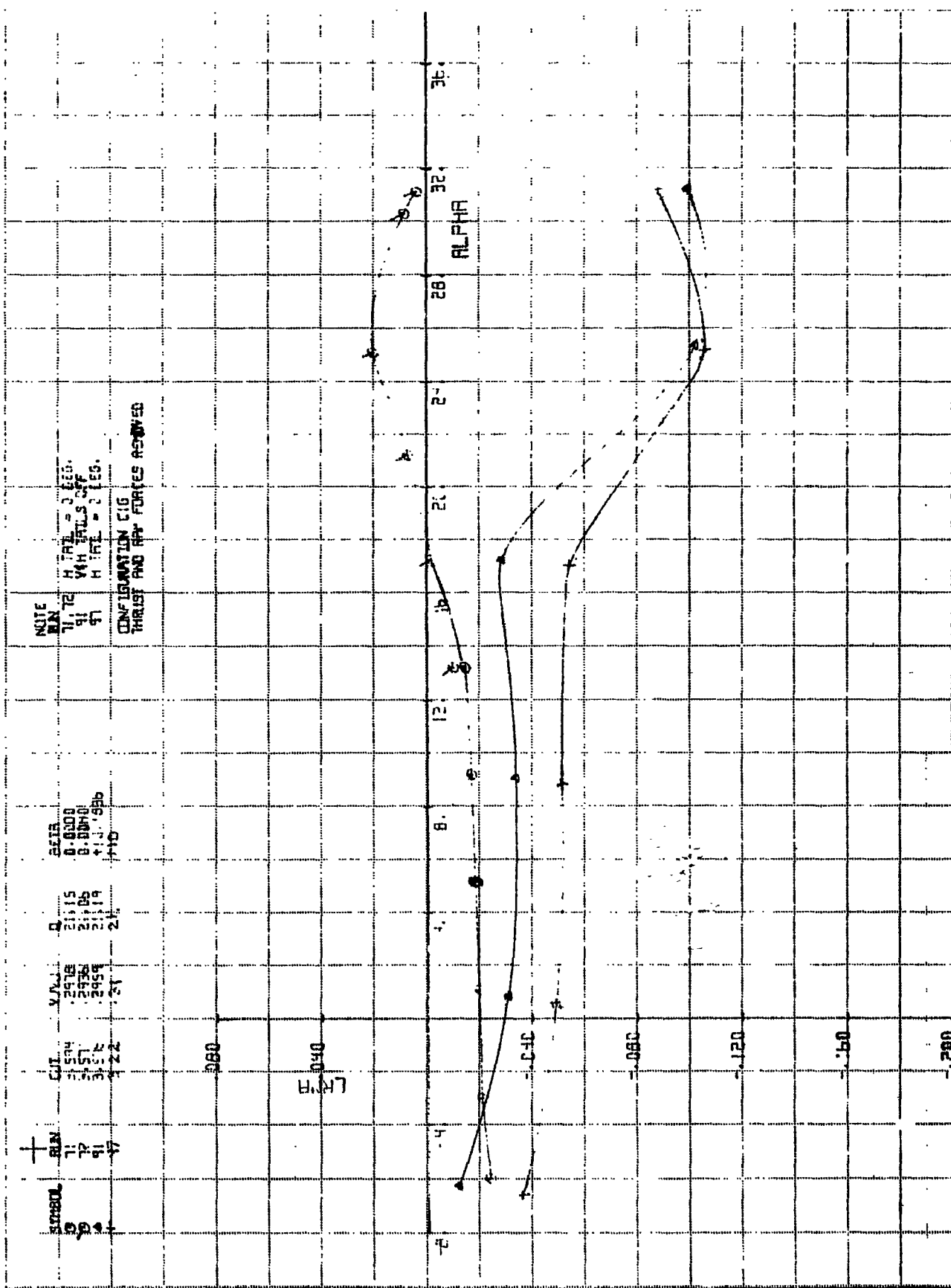
THE BOEING COMPANY

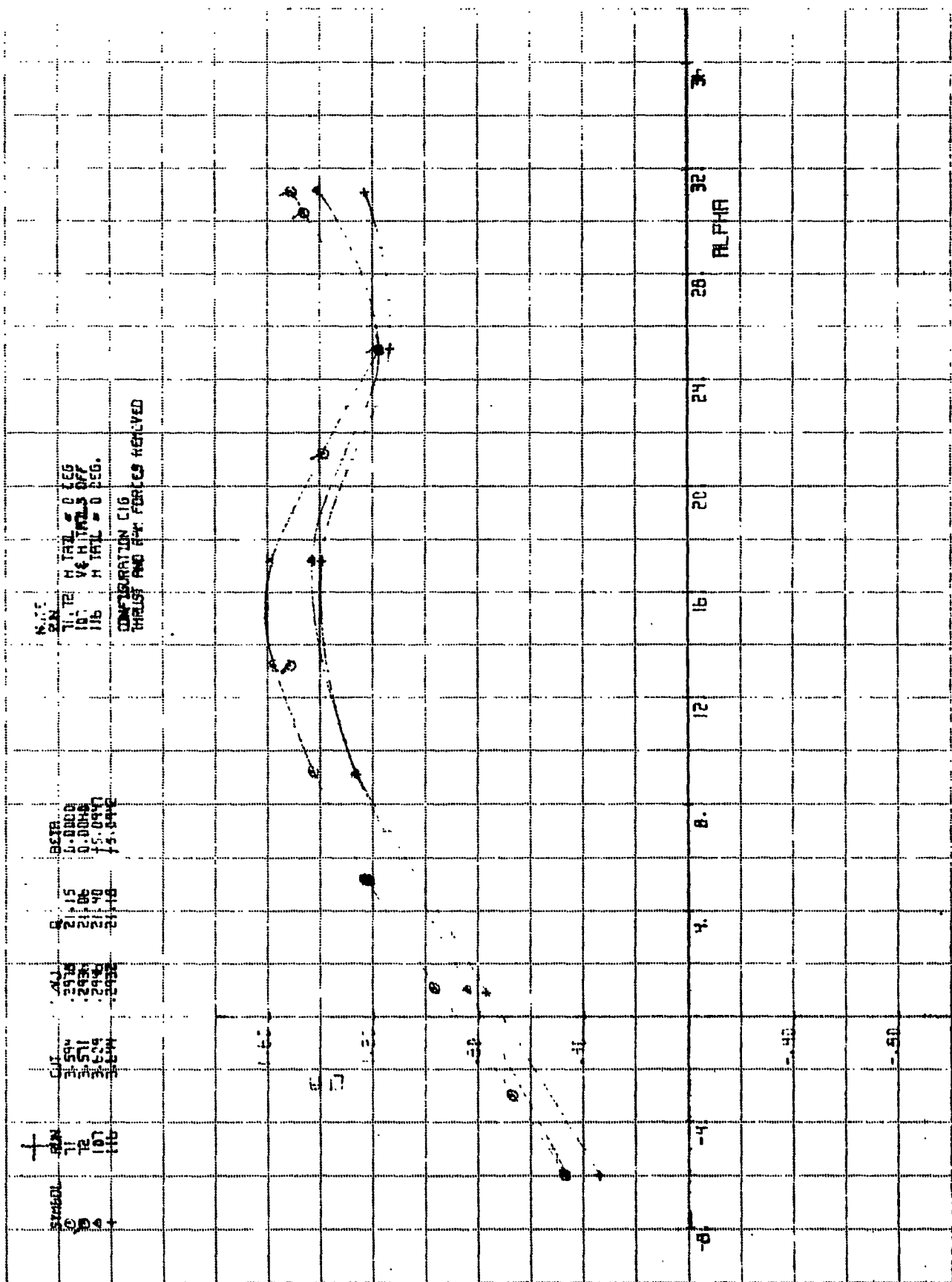
LSHTB 2541

RM - 1354

FIG. A.14
CONT'D

PAGE
166





NITE
 11.12 H TAIL = 0.55
 10.1 V4 H TAIL = 0.57
 11.6 H TAIL = 0.56
 CONFIGURATION C16
 THRUST AND RW FORCES REMOVED

BCTR
 0.0000
 0.0048
 0.0447
 15.842

21.15
 21.06
 21.40
 21.18

2978
 2934
 2946
 2932

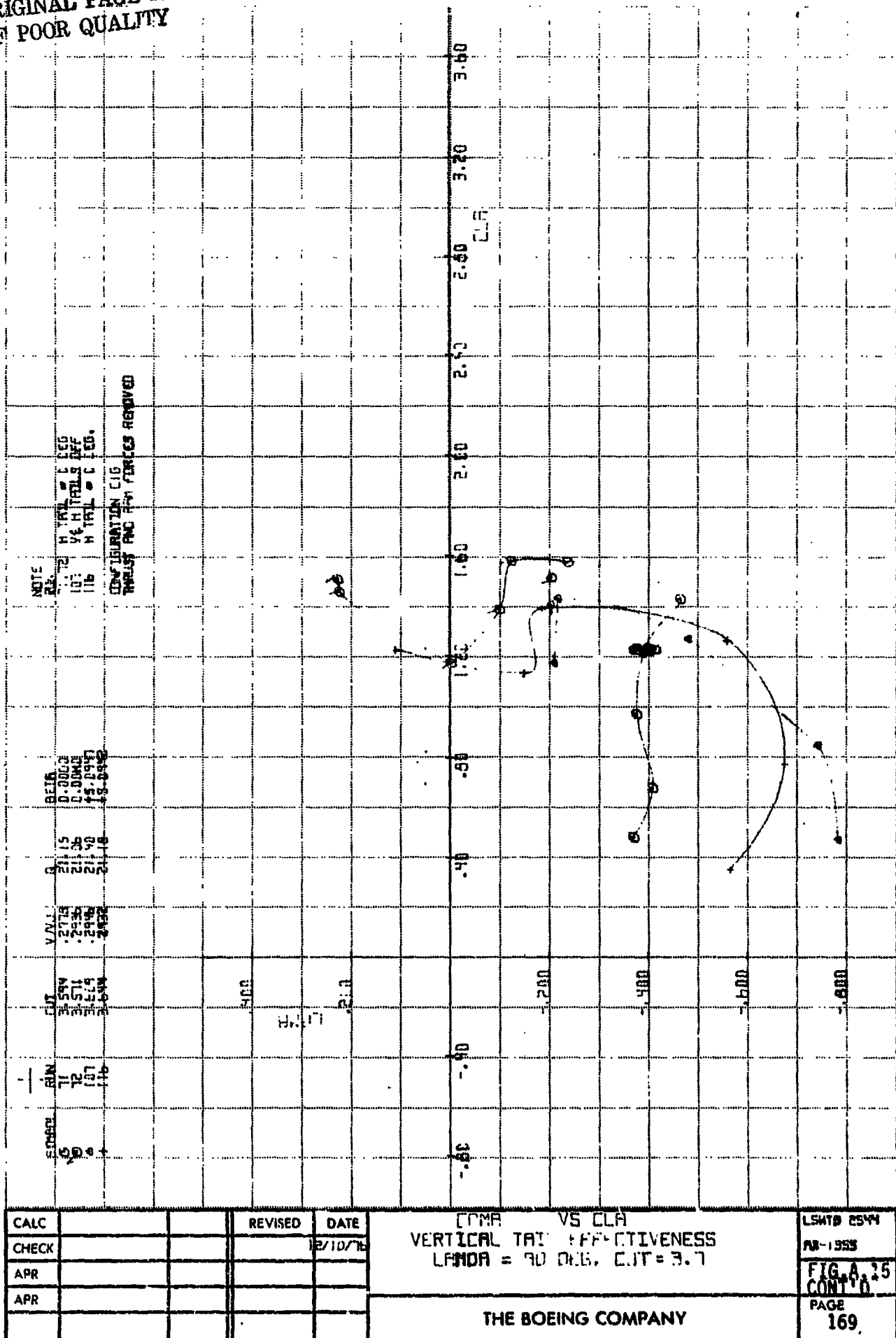
3.571
 3.571
 3.571
 3.571

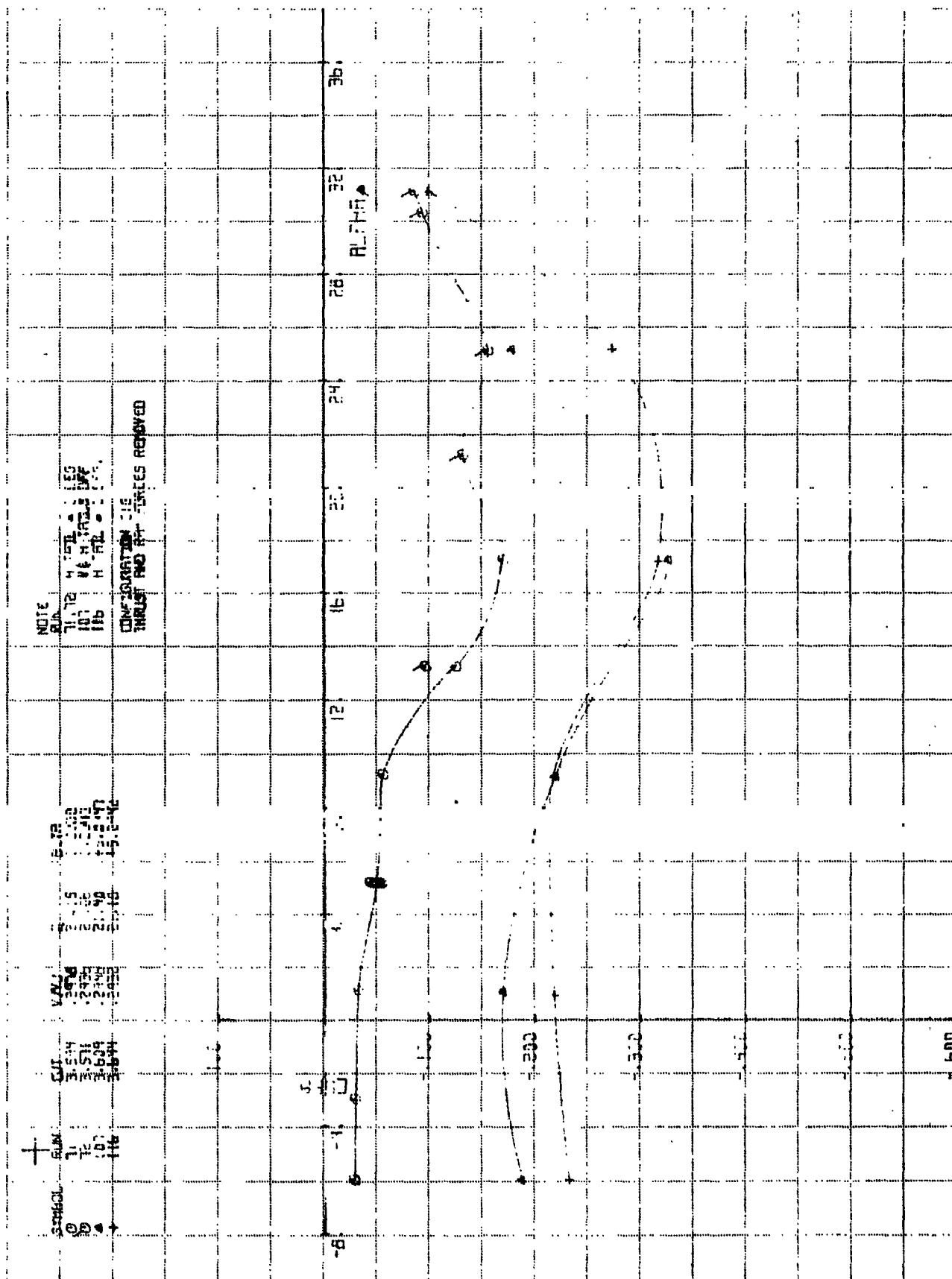
11
 12
 107
 116

CALC		REVISED	DATE	CL vs ALPHA VERTICAL TAIL EFFECTIVENESS LAMDF = 90 DEG, COT = 3.1	LSMTB 2544
CHECK			2/10/78		AX-1355
APR					FIG. A. 15
APR					PAGE 167
THE BOEING COMPANY					

FIG 3
 7.

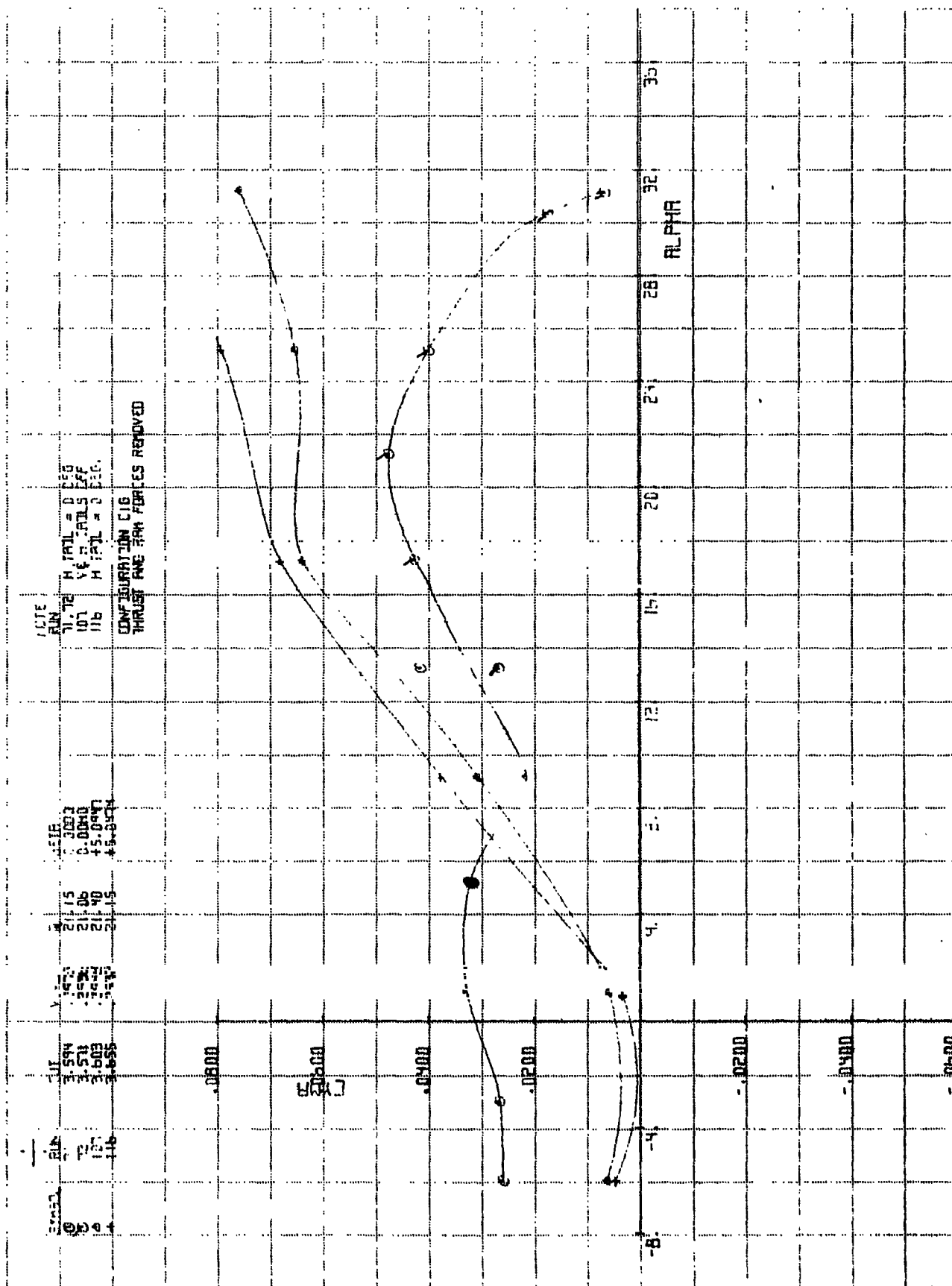
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OF POOR QUALITY



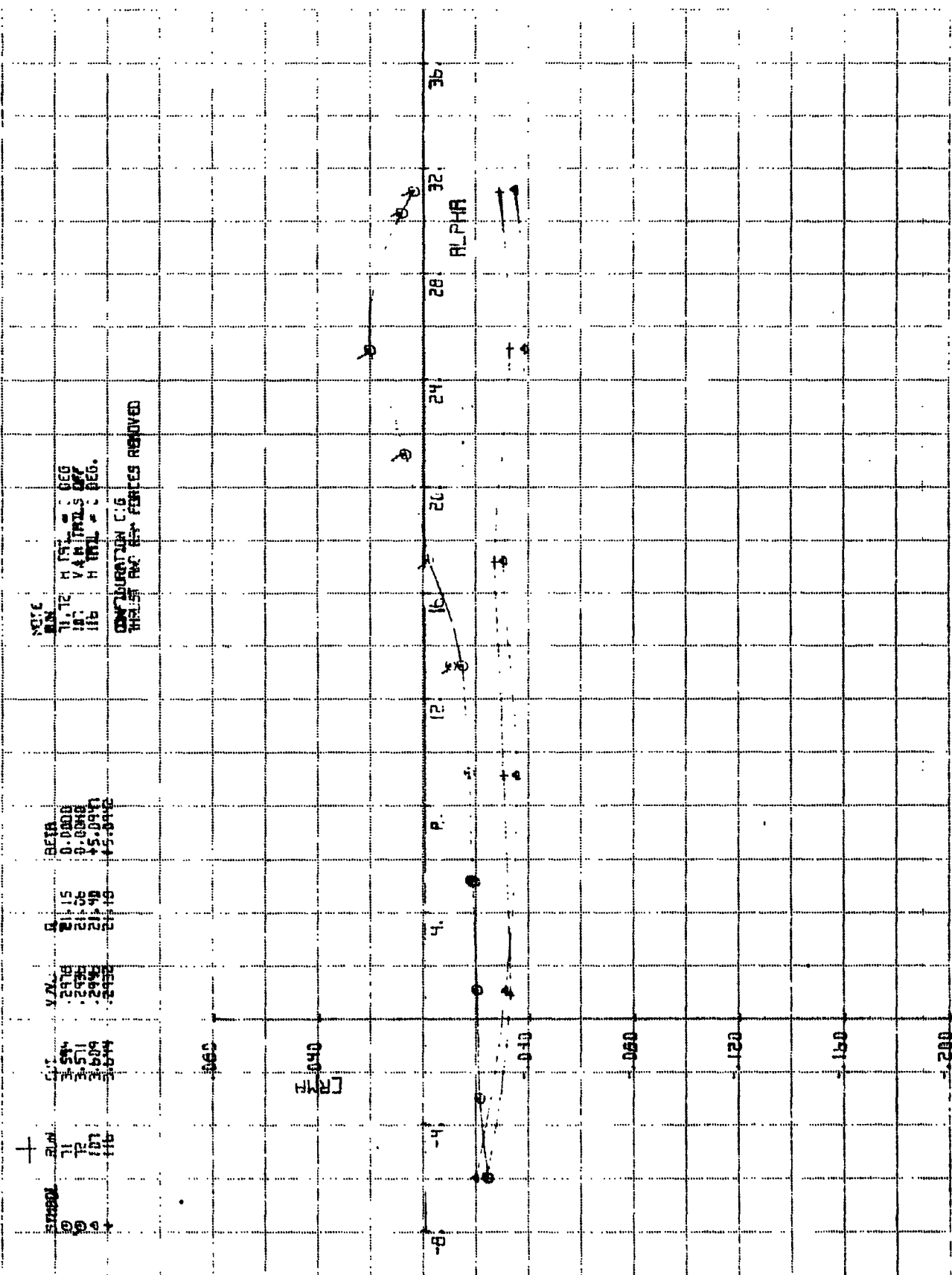


NOTE: CL/alpha = 0.112
CL/alpha = 0.107
CL/alpha = 0.106
CL/alpha = 0.105
CL/alpha = 0.104
CL/alpha = 0.103
CL/alpha = 0.102
CL/alpha = 0.101
CL/alpha = 0.100
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CL/alpha = 0.068
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CL/alpha = 0.063
CL/alpha = 0.062
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CL/alpha = 0.025
CL/alpha = 0.024
CL/alpha = 0.023
CL/alpha = 0.022
CL/alpha = 0.021
CL/alpha = 0.020
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CL/alpha = 0.006
CL/alpha = 0.005
CL/alpha = 0.004
CL/alpha = 0.003
CL/alpha = 0.002
CL/alpha = 0.001
CL/alpha = 0.000

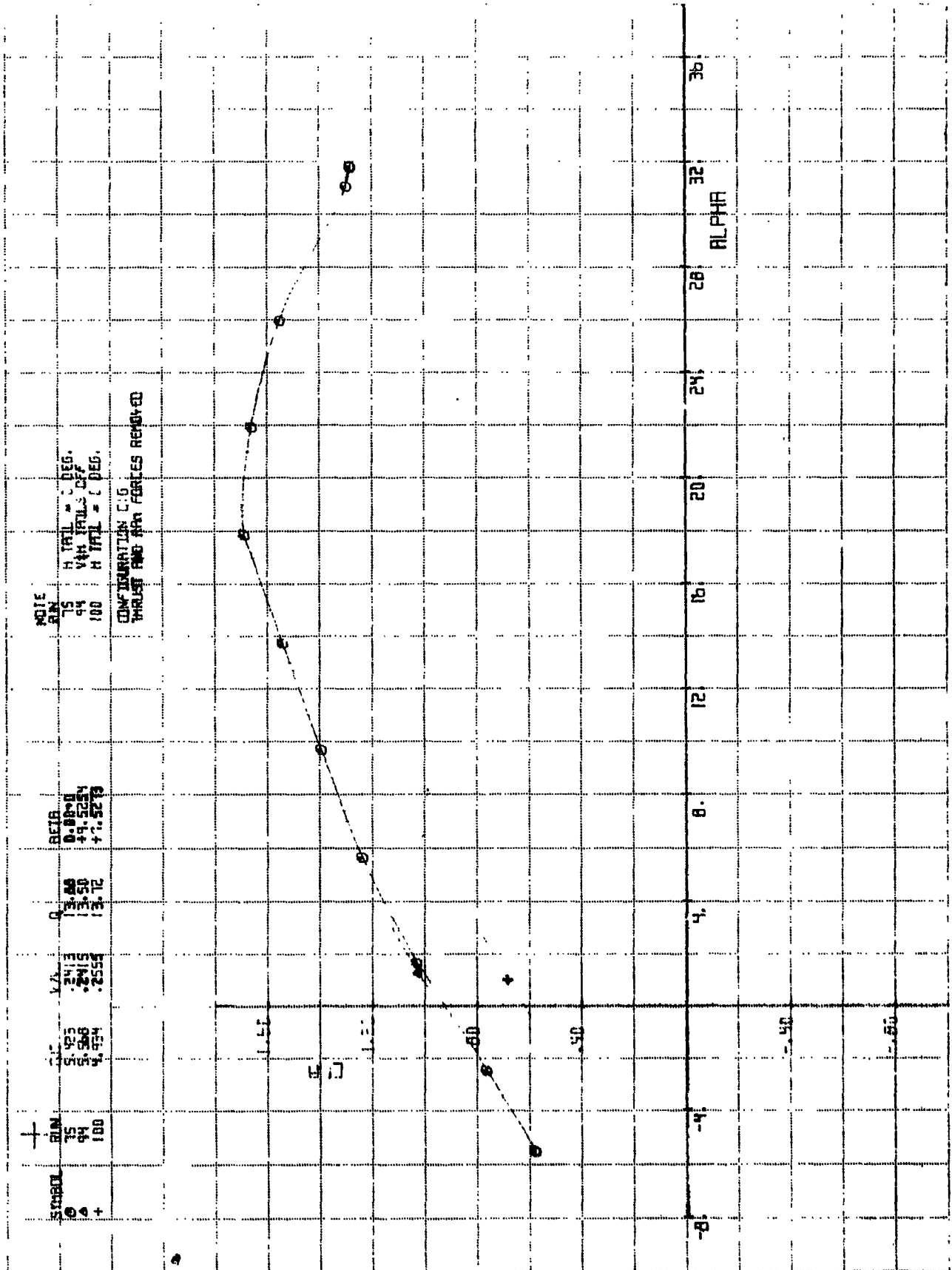
CALC		REVISED	DATE	LIFTING LINE EFFECTIVENESS LAMINAR - NO DEG. LUT = 3.7 THE BOEING COMPANY	LEMTO ESTN RX 1355 FIG. A.15 CONT'D PAGE 170
CHECK			7/10/16		
APR					
APR					



CALC			REVISED	DATE	1/19/77	ALPHA	CL
CHECK							
APR							
APR							
THE BOEING COMPANY						LSMTB 25-11	171

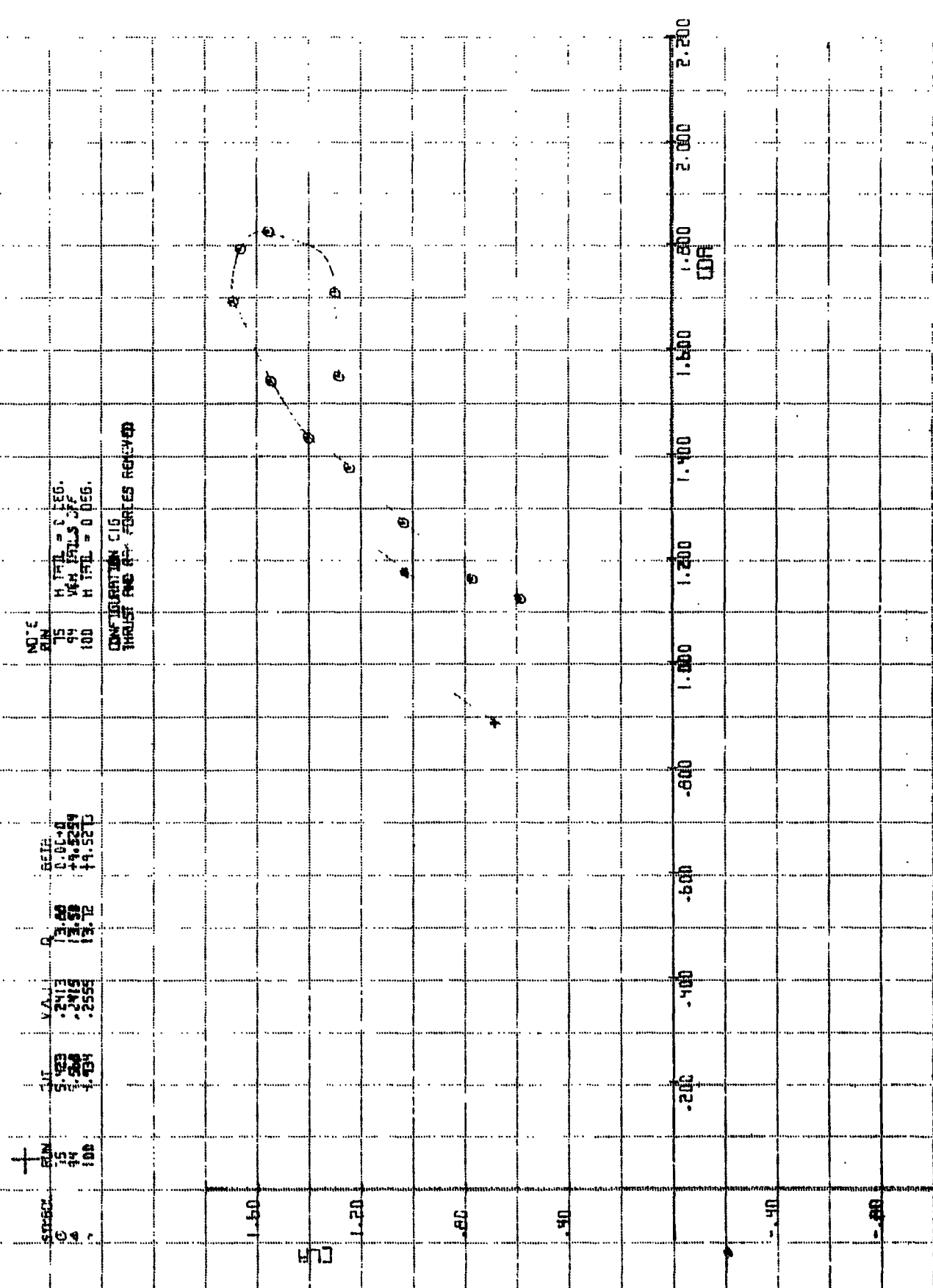


CALL:		REVISED	DATE	CRMA VS ALPHA VERTICAL THRUST EFFECTIVENESS LAMBDA = 90 DEG, CUT = 3.1 THE BOEING COMPANY	LSWTB 25-4 RA-1955 FIG. A.15 CONT'D PAGE 172
CHECK:			2/10/78		
APR					
APR					

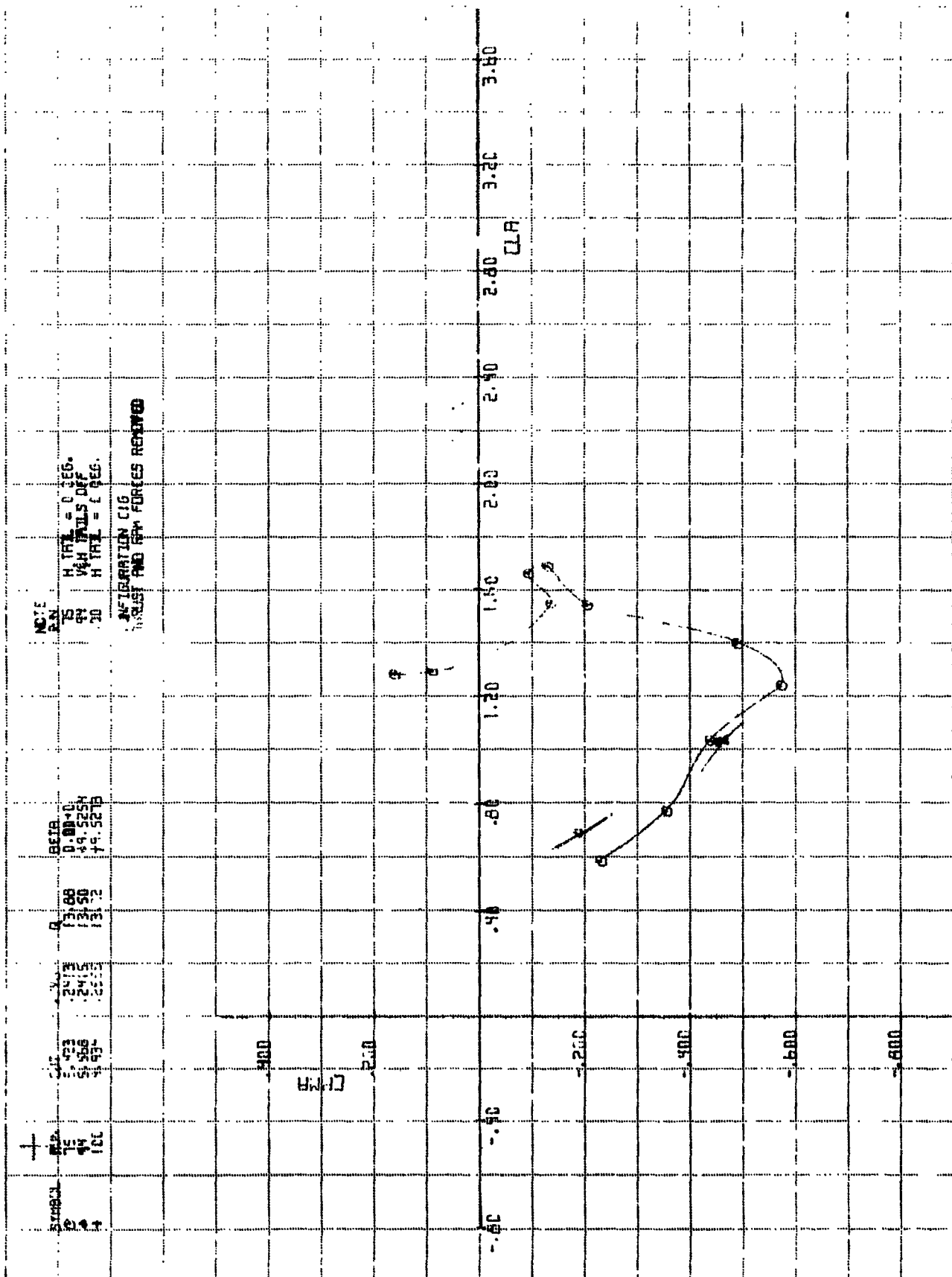


CALC			REVISED	DATE	CL vs ALPHA VERTICAL TAIL EFFECTIVENESS LAMDA = .90 DEG. C.T. = 5.5	LSMTB 25-41
CHECK				2/10/76		RA-1355
APR						FIG.A.16
APR						PAGE 173
THE BUEING COMPANY						

F16 33
13.



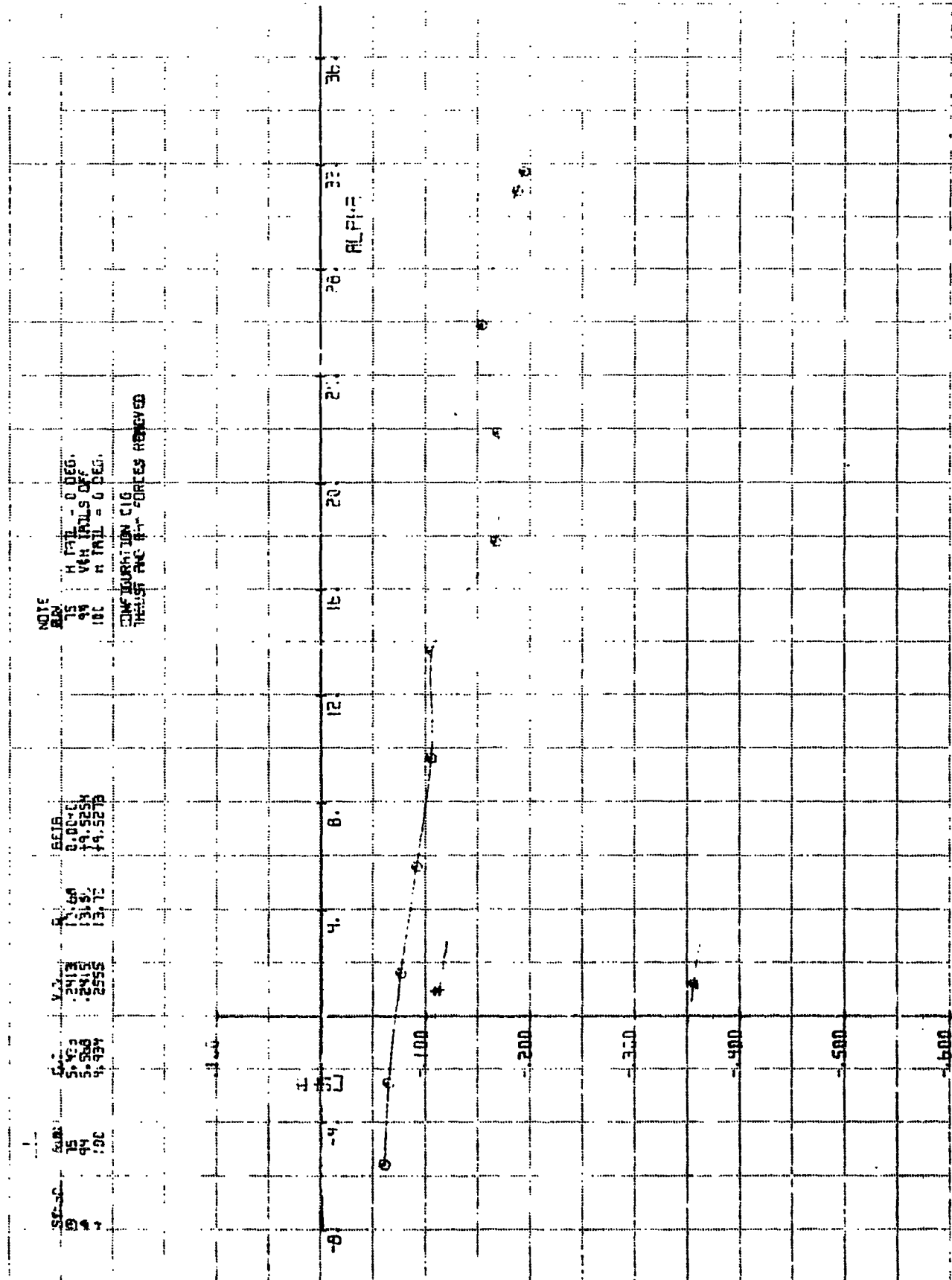
CALC		REVISED	DATE	CLA VS CDR	LSMTB 2544
CHECK			2/10/72	VERTICAL TAIL EFFECTIVENESS	FX-1335
APR				LAMBDA = 90 DEG, CJT = 5.5	FIG. A.16
APR					CONT'D
				THE BOEING COMPANY	PAGE 174



NOTE
 1. H. TAIL = 0.165.
 2. V. H. TAIL OFF
 3. H. TAIL = 0.165.
 4. ACCELERATION CIG
 5. H. TAIL AND H. TAIL FORCES REMOVED

CALC			REVISED	DATE	CPMA VS C/A	LSMTB 2544
CHECK				2/10/7	VERTICAL TAIL EFFECTIVENESS	AN-1393
APR					LHMTA = 90 DEG. CUT = 5.5	FIG. A. 16
APR						CONT'D
					THE BOEING COMPANY	PAGE 175

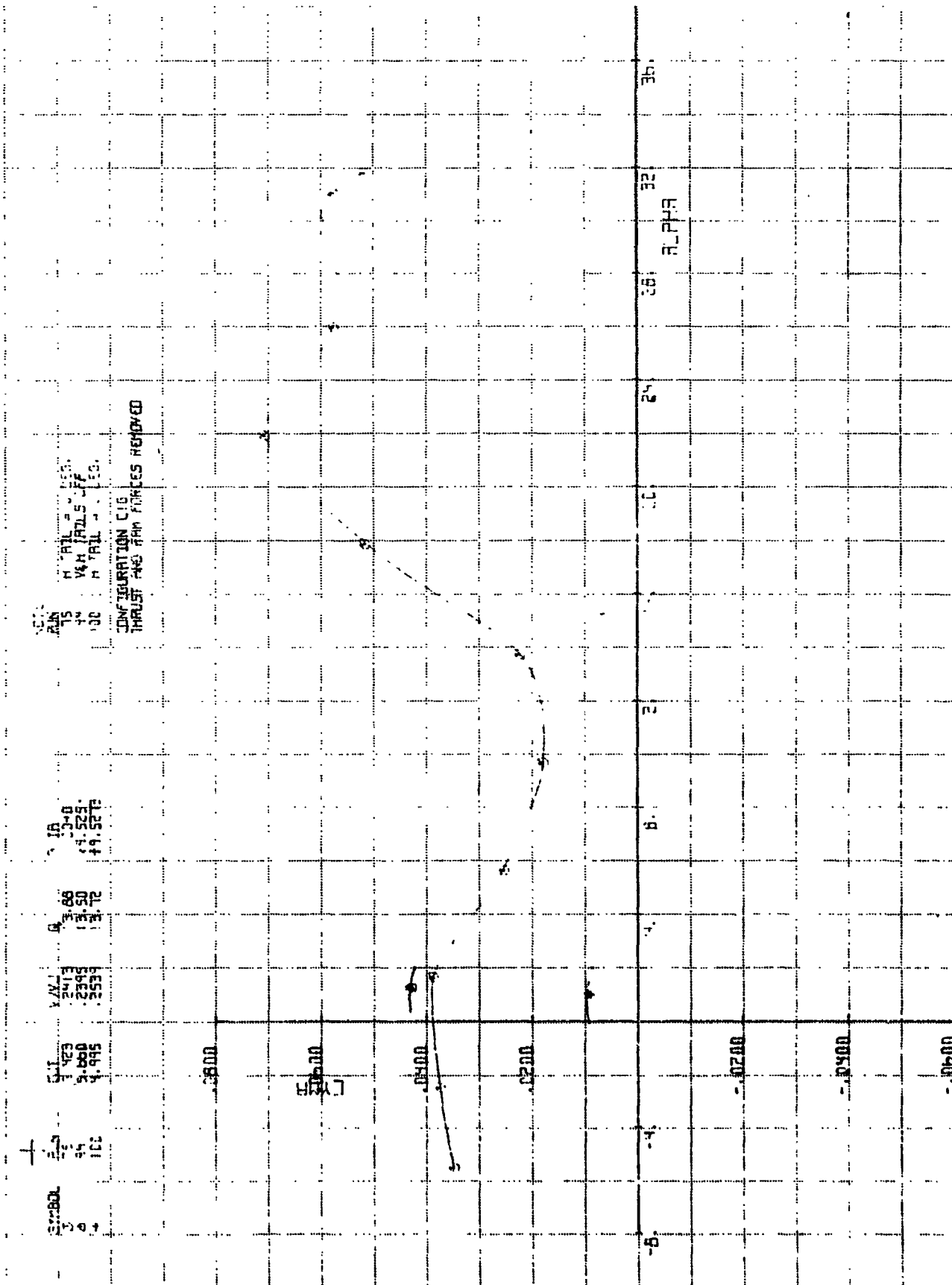
FIG. 11
 II.



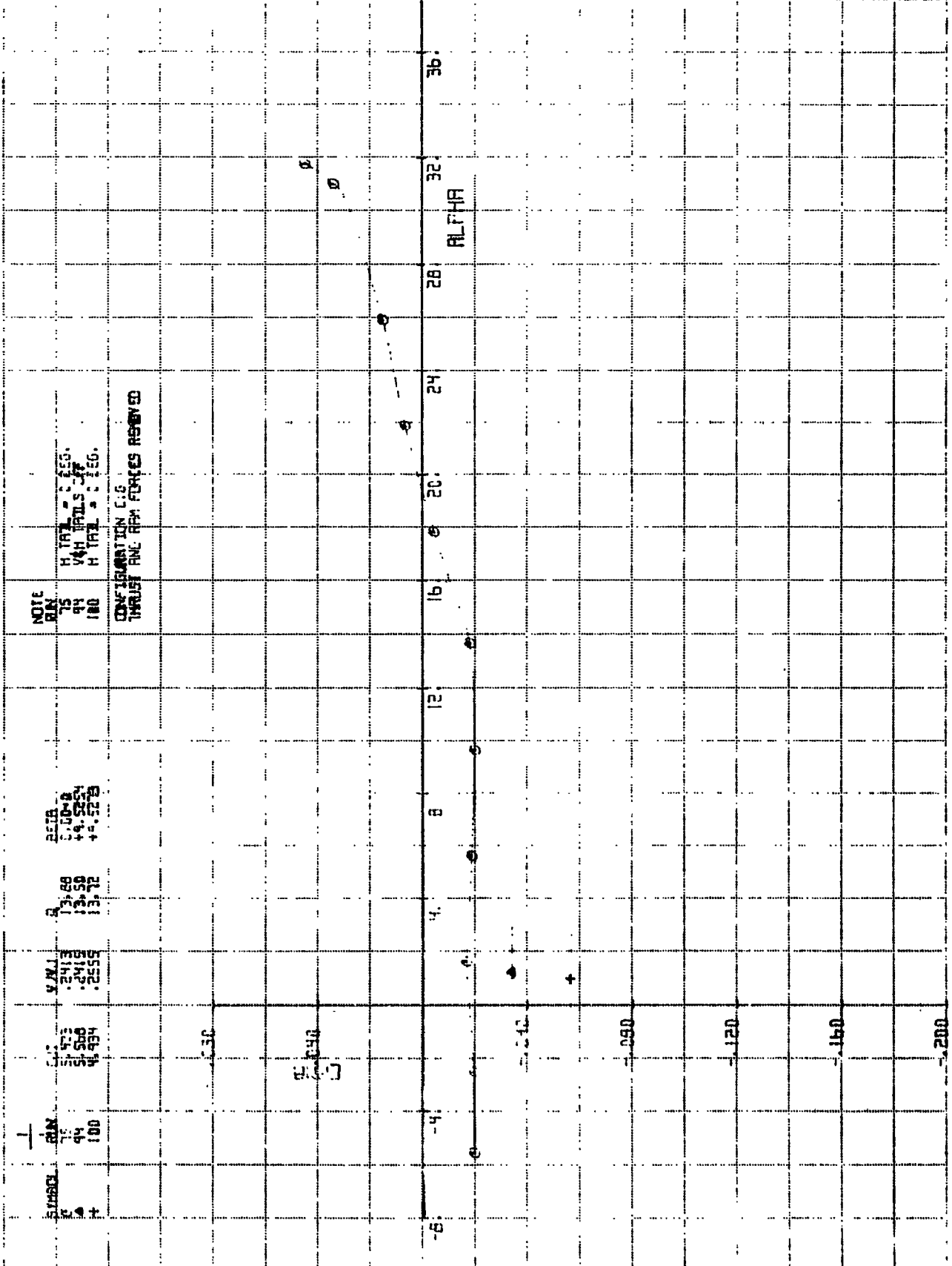
CALC		REVISED	DATE	2/10/78	CSFH V-ALPHA	LSHTR 2544
CHECK					VERTICAL THRUST EFFECTIVENESS	AX-1355
APP					LHMR = 90 DEG. C/T = 5.5	FIG. A.16
APP						CONT'D
					THE BOEING COMPANY	PAGE 176

FL 33
13.

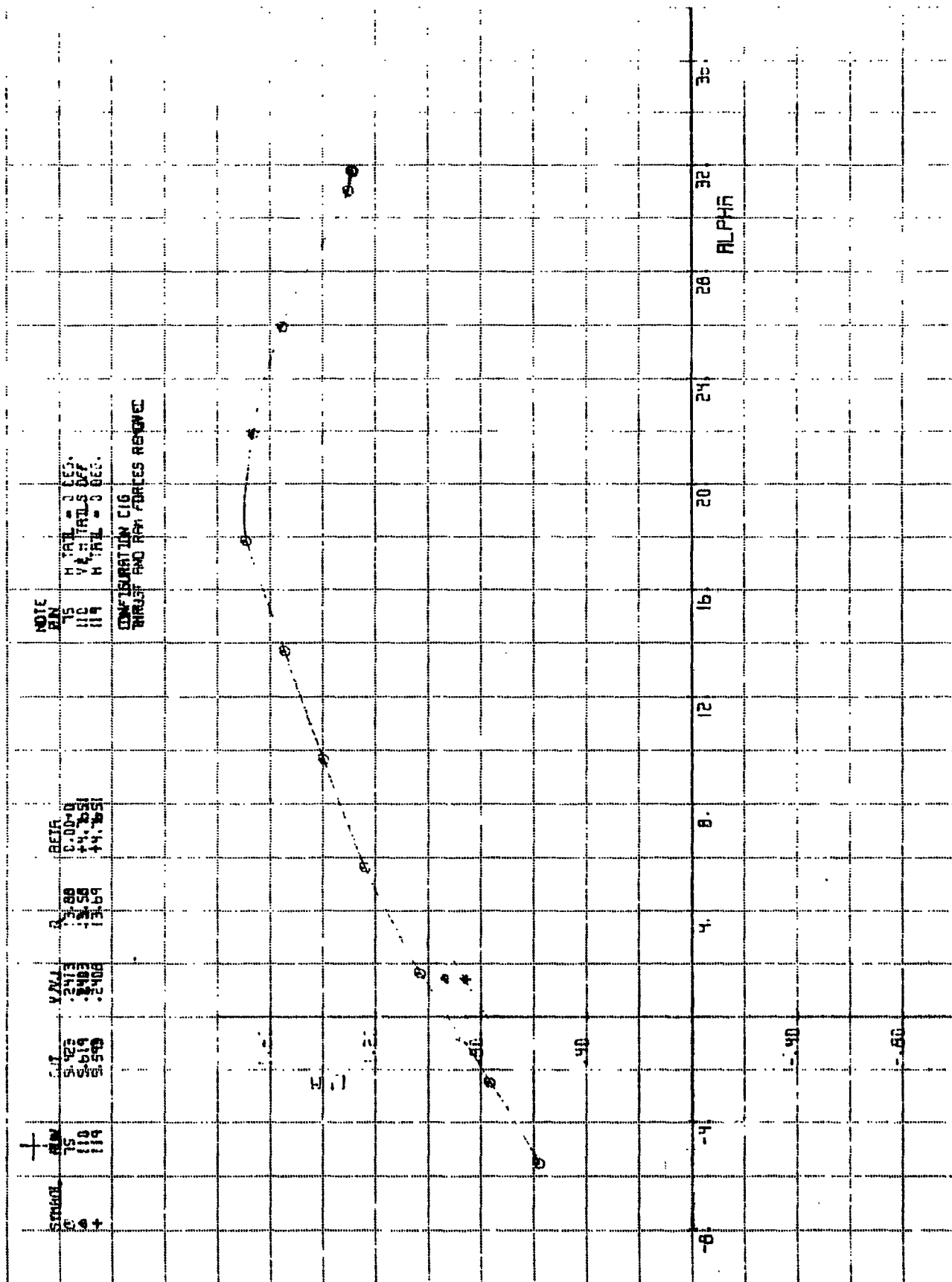
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CALC		REVISED	DATE	CLYMR VS ALPHA VERTICAL TAIL EFFICIENCY ALPHA = 40 DEG. C.T. = 0.5	LSHTB 2541
CHECK			01/19/71		RX-1333
APR					FIG. A.16
APR					CONT'D
				THE BOEING COMPANY	PAGE 17



CALC		REVISED	DATE	LAMBDA VS. F VERTICAL THRUST EFFECTIVENESS LAMBDA = 90 DEG. LOT = 5.0	LSWT 2541 RX-1355 FIG. A.16 CONT'D PAGE 178
CHECK			2/10/76		
APR					
APR					
				THE BOEING COMPANY	

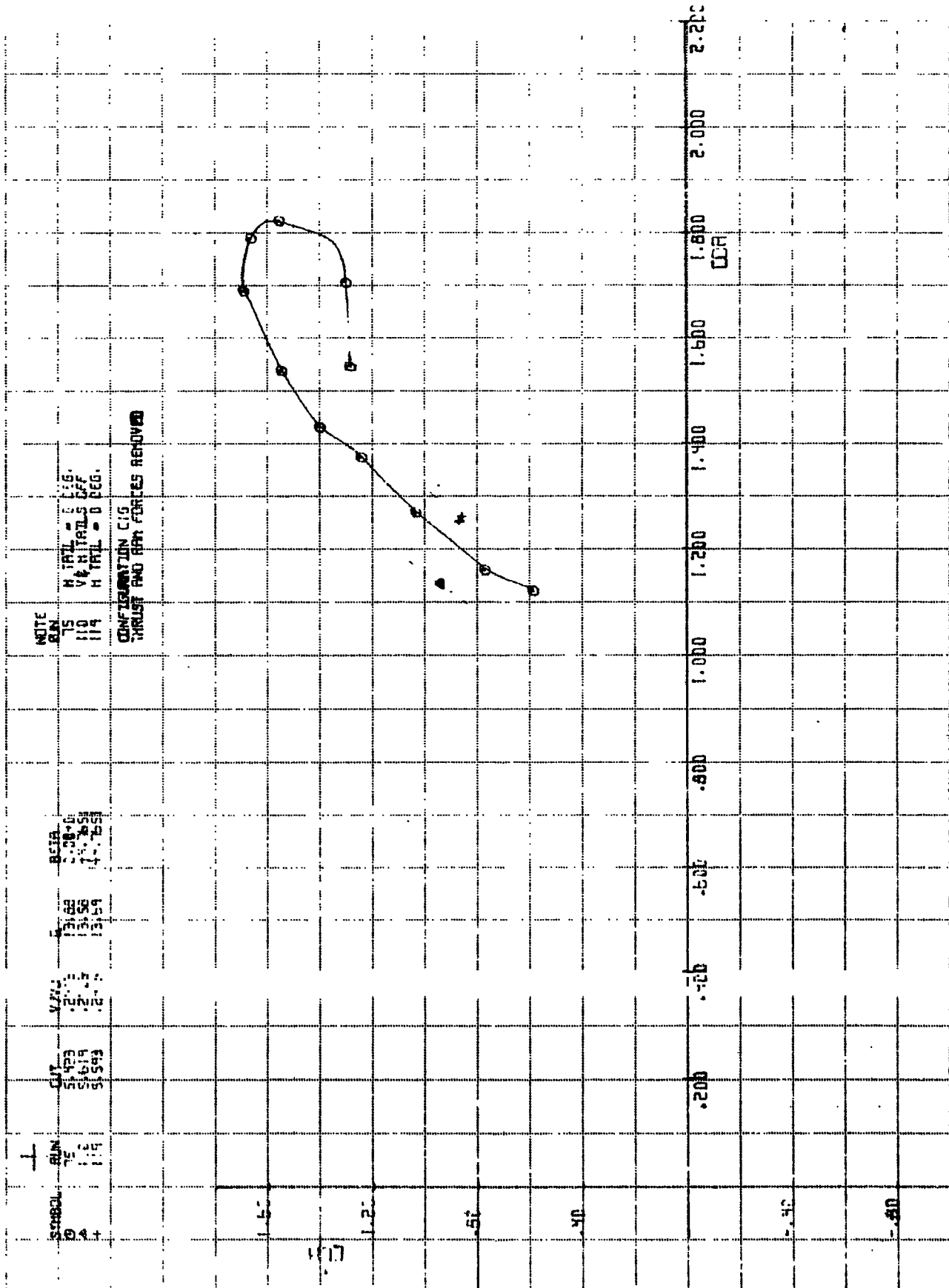


CALC			REVISED	DATE	CIG VS ALPHA VERTICAL ANGLE EFFECTIVENESS LAMBDA = NO PIG. CUT = 5.1"	LSMTD 25441
CHECK				2.10.76		HX 1355
APR						FIG.A. 17
APR						GE 179
						THE BOEING COMPANY

CALC		REVISED	DATE
CHECK			2/10/78
APR			
APR			

CLH VC LDR
 VERTICAL TAIL EFFECTIVENESS
 LHMDH = 9.0 DEG. CUT = 5.5

LSMTB 2544
 RX-1355
 FIG. A. 17
 CONT'D
 PAGE
 180

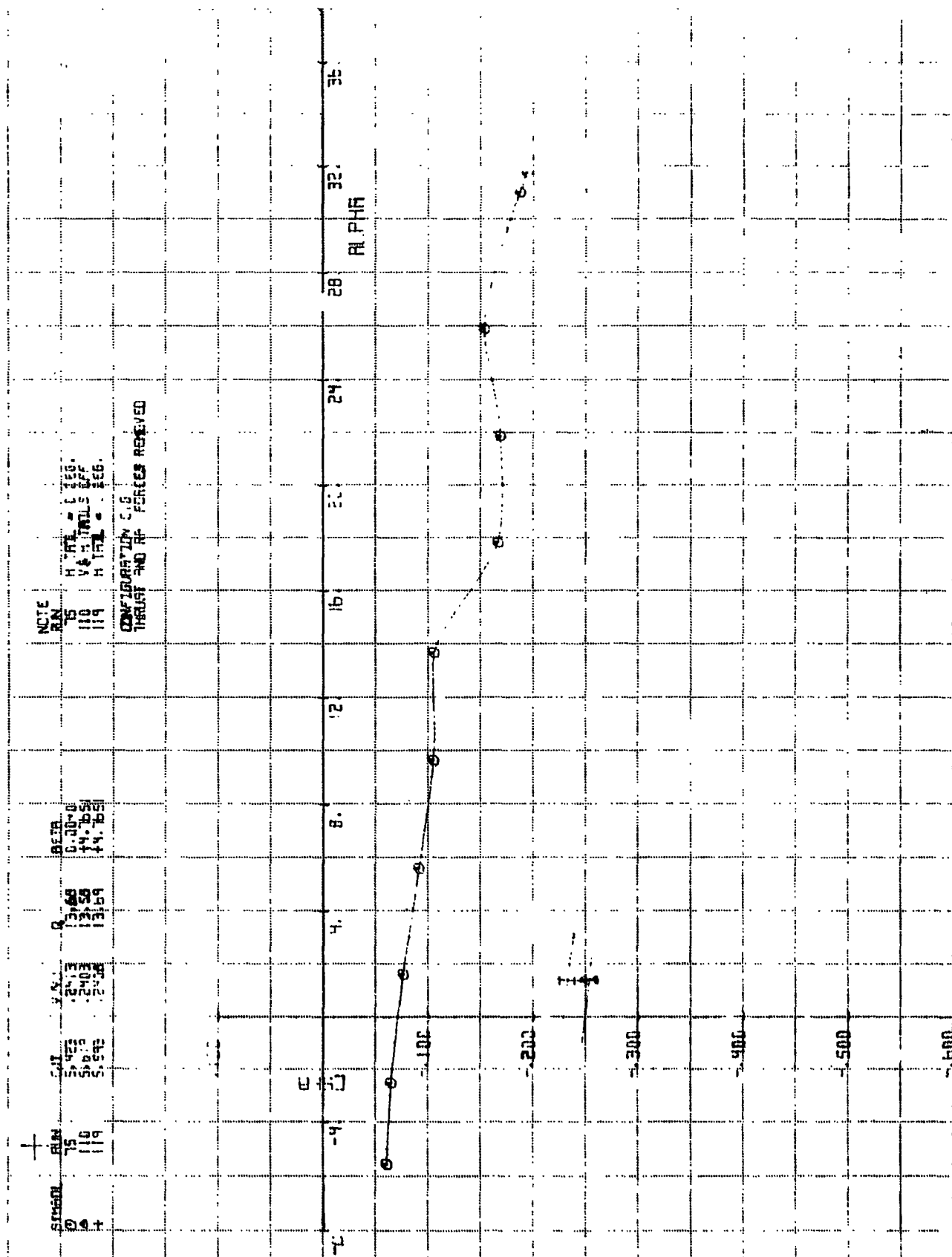


CALC			REVISED	DATE
CHECK				7/10/78
APR				
APP				

25FR V: 1 PHH
 VERTICAL TAIL EFFECTIVENESS
 LAMDA = 90 DEG. C/T = 5.5

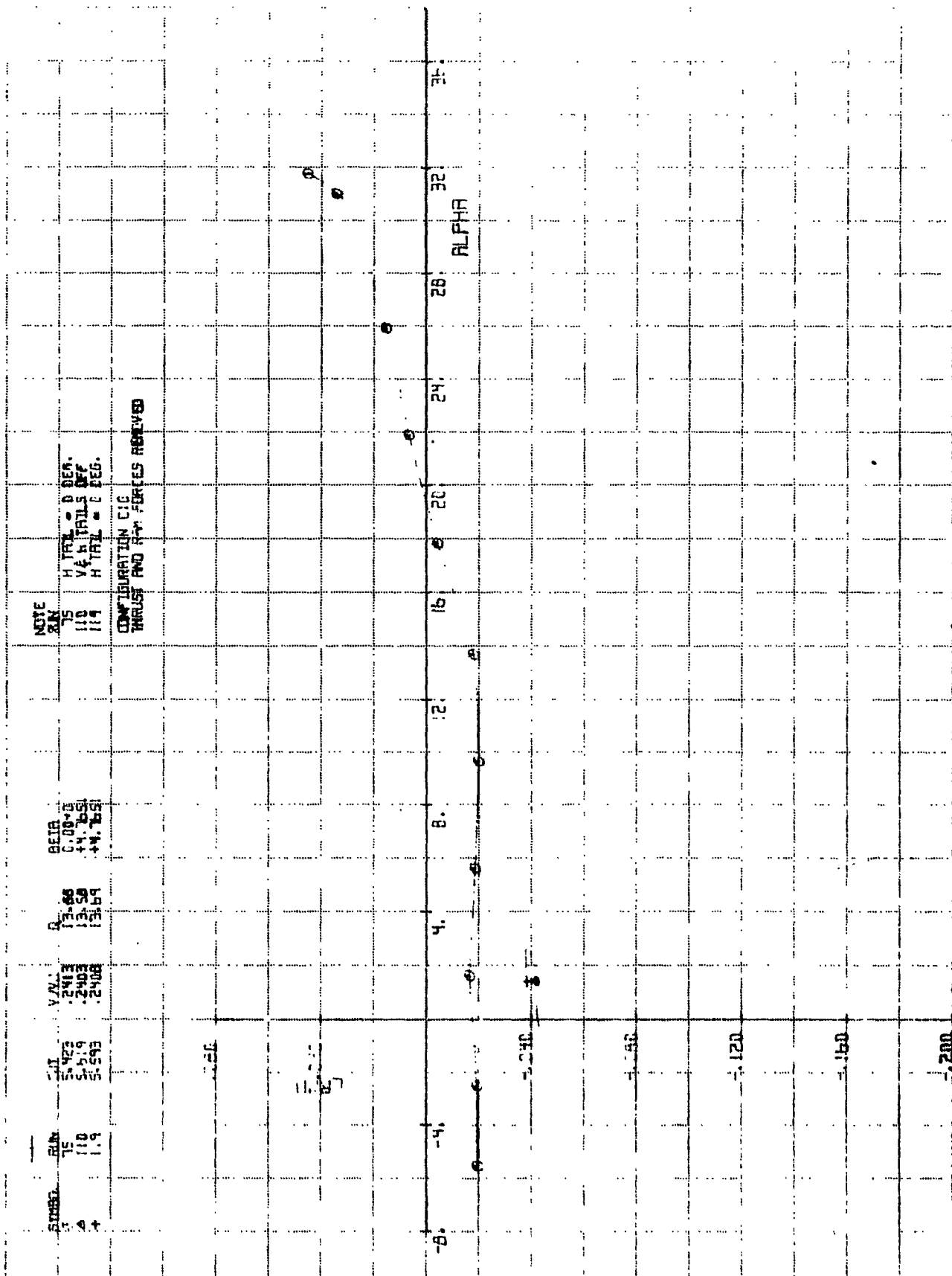
LSWTR 2544
 RX-1355
 FIG.A.17
 CONT'D
 PAGE
 182

THE BOEING COMPANY

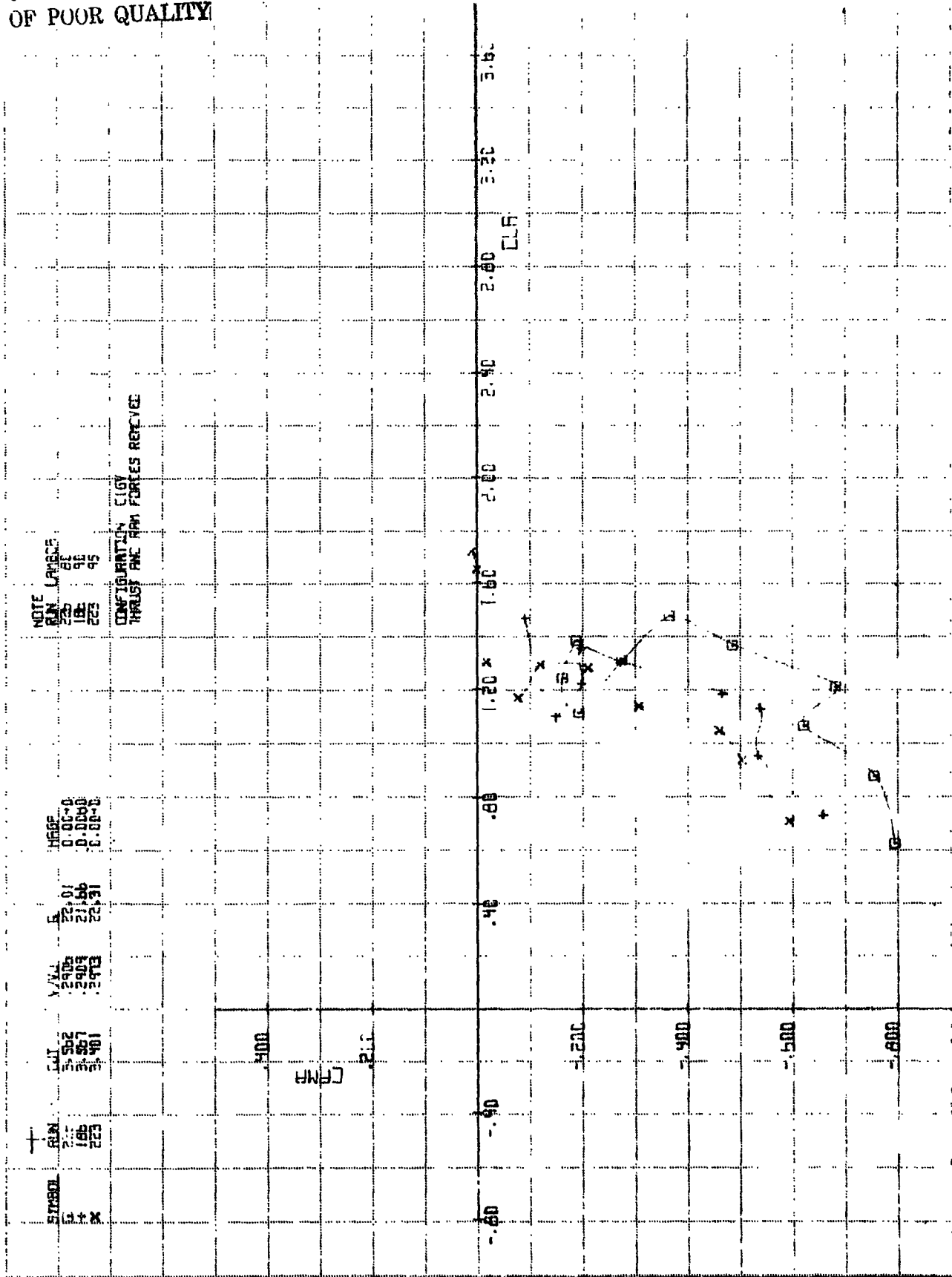


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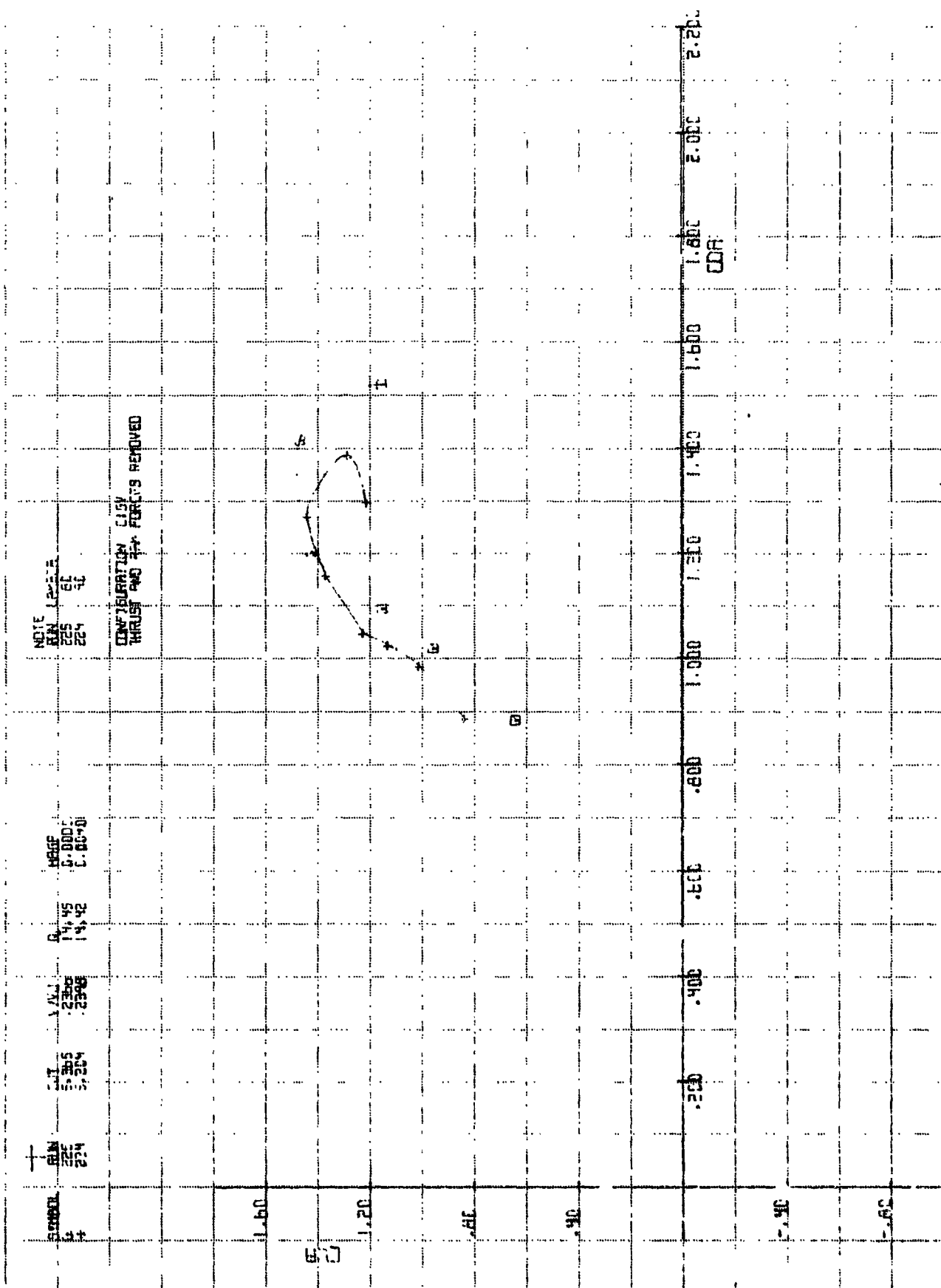
CALC			REVISED	DATE	LYME V. ALPHA	LSMTB 2544
CHECK				3/19/77	VERTICAL TAIL EFFECTIVENESS	RX-1355
APR					LAMDA = 90 DEG. CUT = 5.5	FIG. A.17
APR						CONT'D
					THE BOEING COMPANY	PAGE 183



CALC			REVISED	DATE	CRMA VS ALPHA VERTICAL TAIL EFFECTIVENESS LAM. A = 90 DEG. CUT = 5.5	LAM. A 2544
CHECK				2/10/76		AX-1355
APR						FIG. A.17
APR						CONT'D
						PAGE 184
THE BOEING COMPANY						

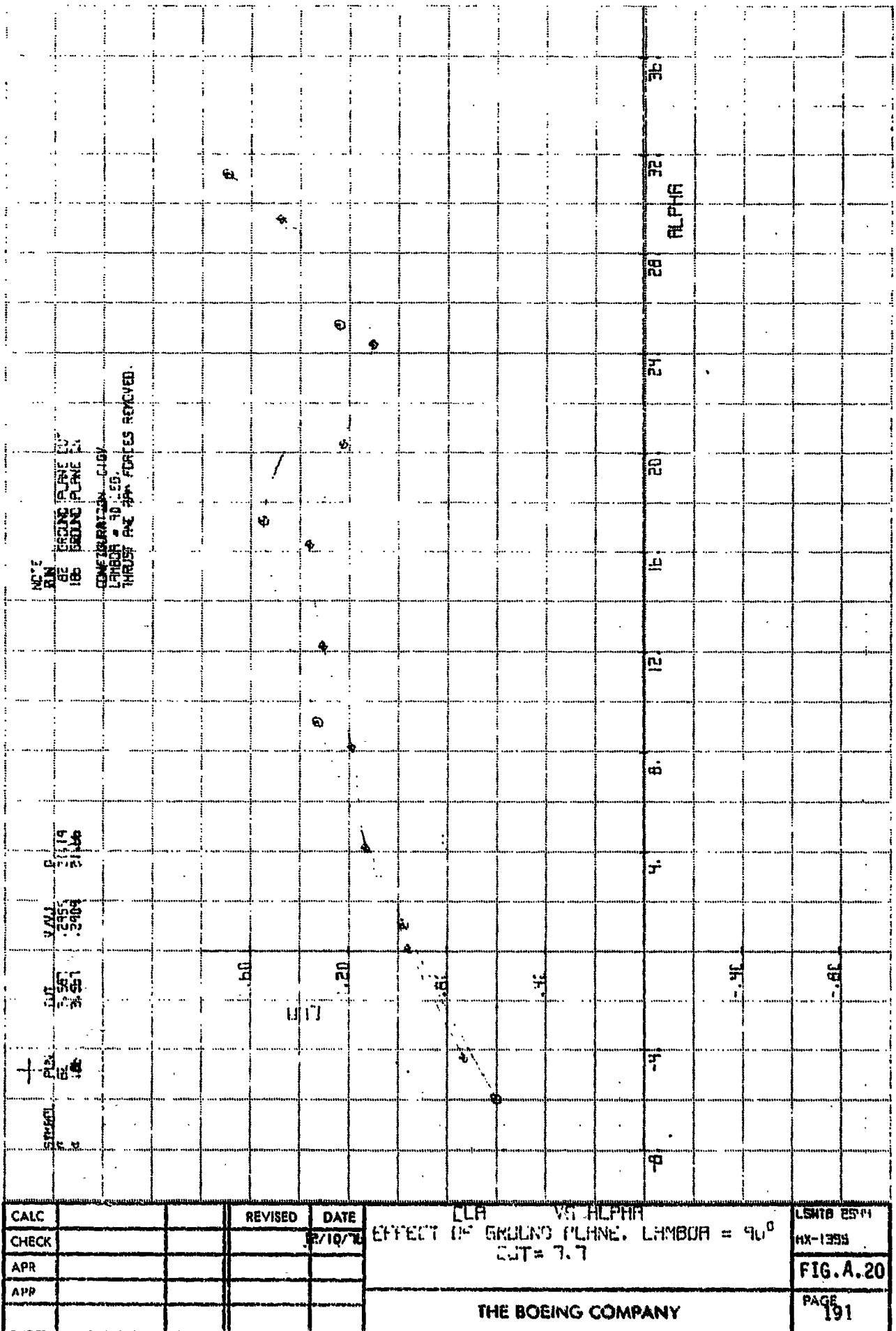


CALC			REVISED	DATE	DATA VS CLR	LSWTS 2544
CHECK				12/17/76	EFFECT OF ENGINE TILT ANGLE	AX-1355
APR					ENT = 3.7. GROUND PLANE INSTALLED	FIG. A.18
APR						CONT'D
					THE BOEING COMPANY	PAGE 187

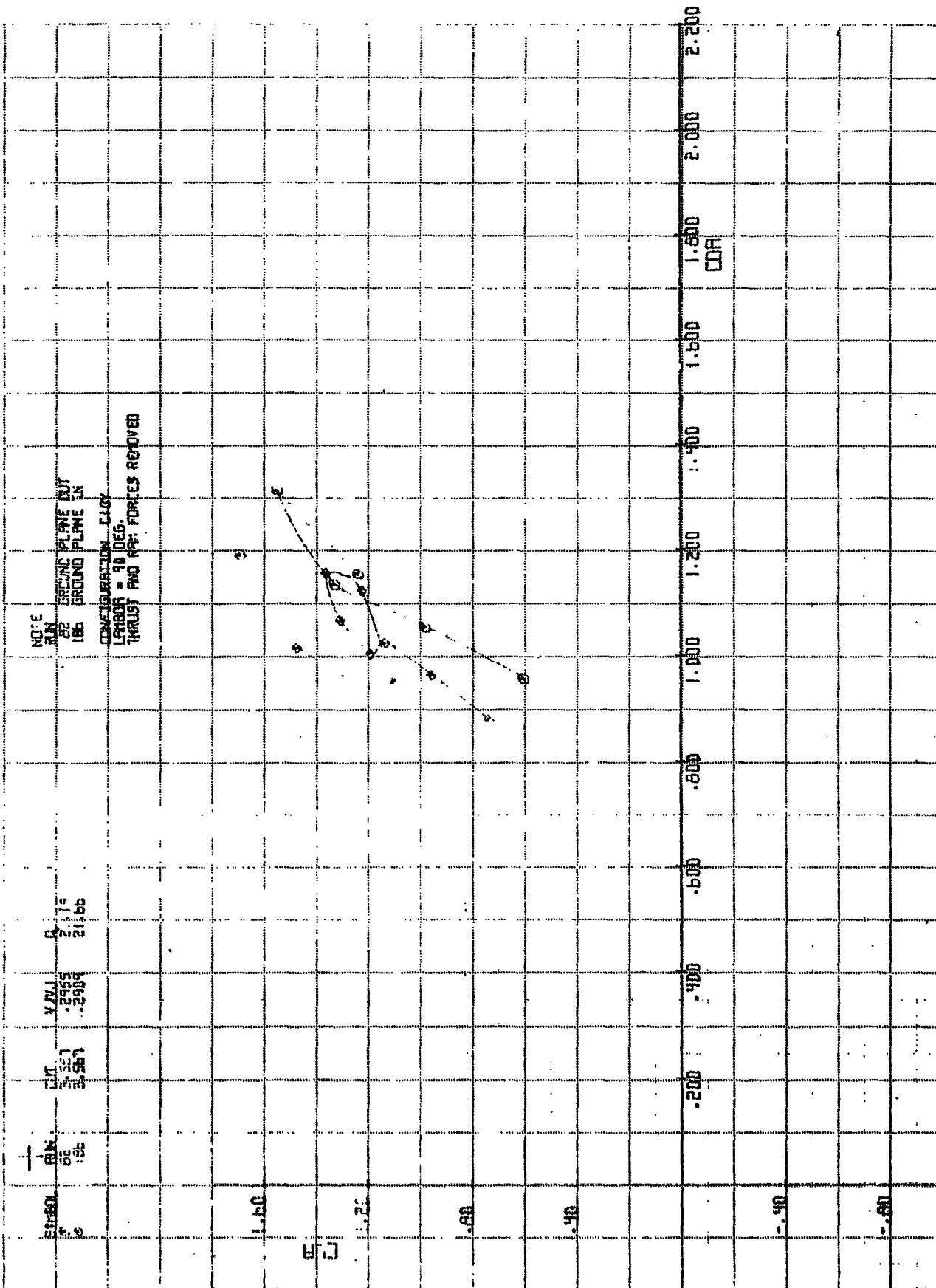


CALC		REVISED	DATE	CLH VS CDH	LGWTS 2544
CHECK			2/10/78	EFFECT OF ENGINE TILT ANGLE	AX-1555
APP				ENT = 1.5, CRUISING PLANE INSTALL.	FIG. A. 19
APR					CONT'D
				THE BOEING COMPANY	PAGE 189

FIG. 19
17.



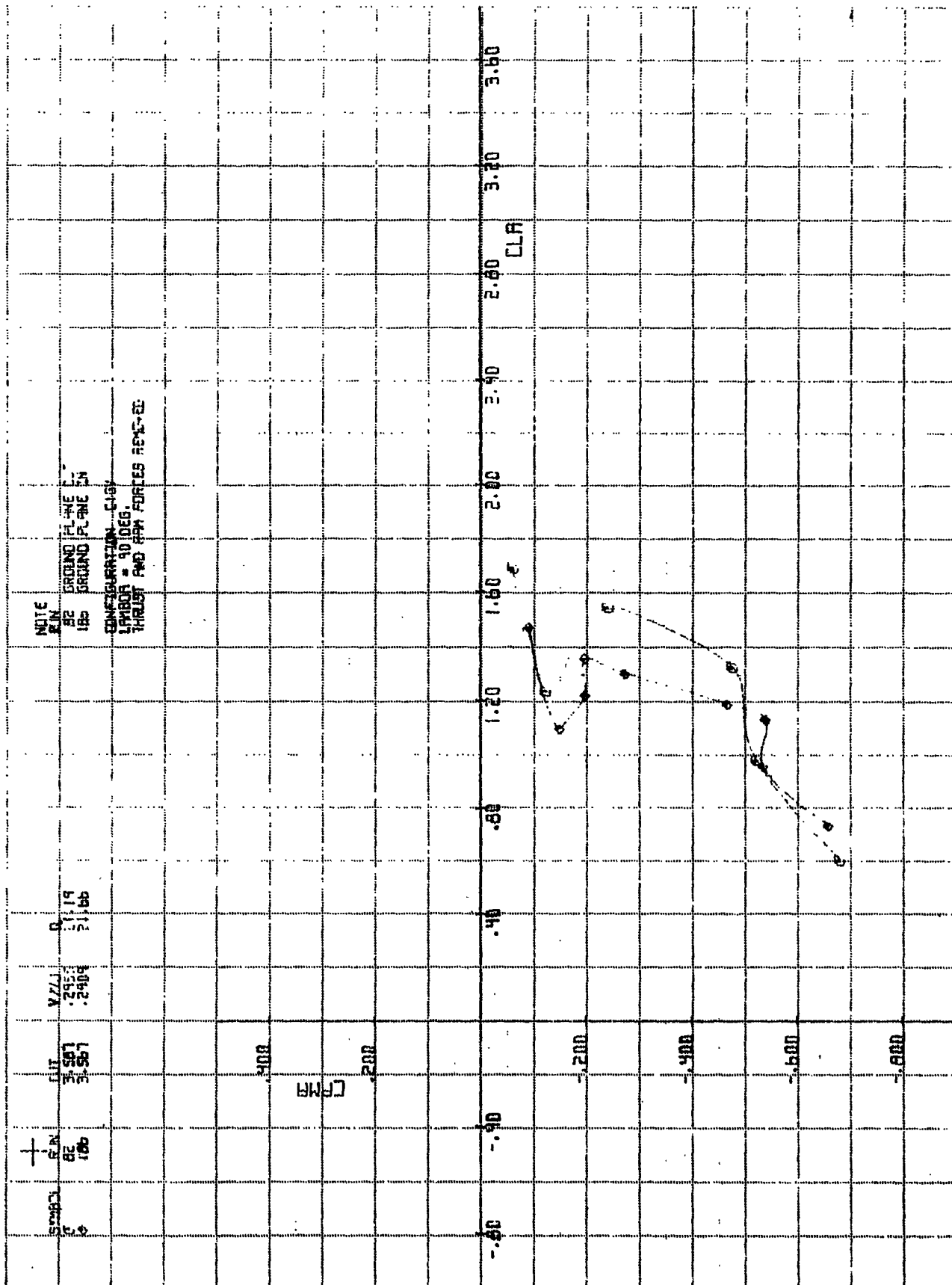
CALC		REVISED	DATE	CL	VS ALPHA	LENB 2541
CHECK			2/10/76	EFFECT OF GROUND PLANE, LAMBDA = 90		MX-1355
APR				CWT = 7.7		FIG. A.20
APP						PAGE 191



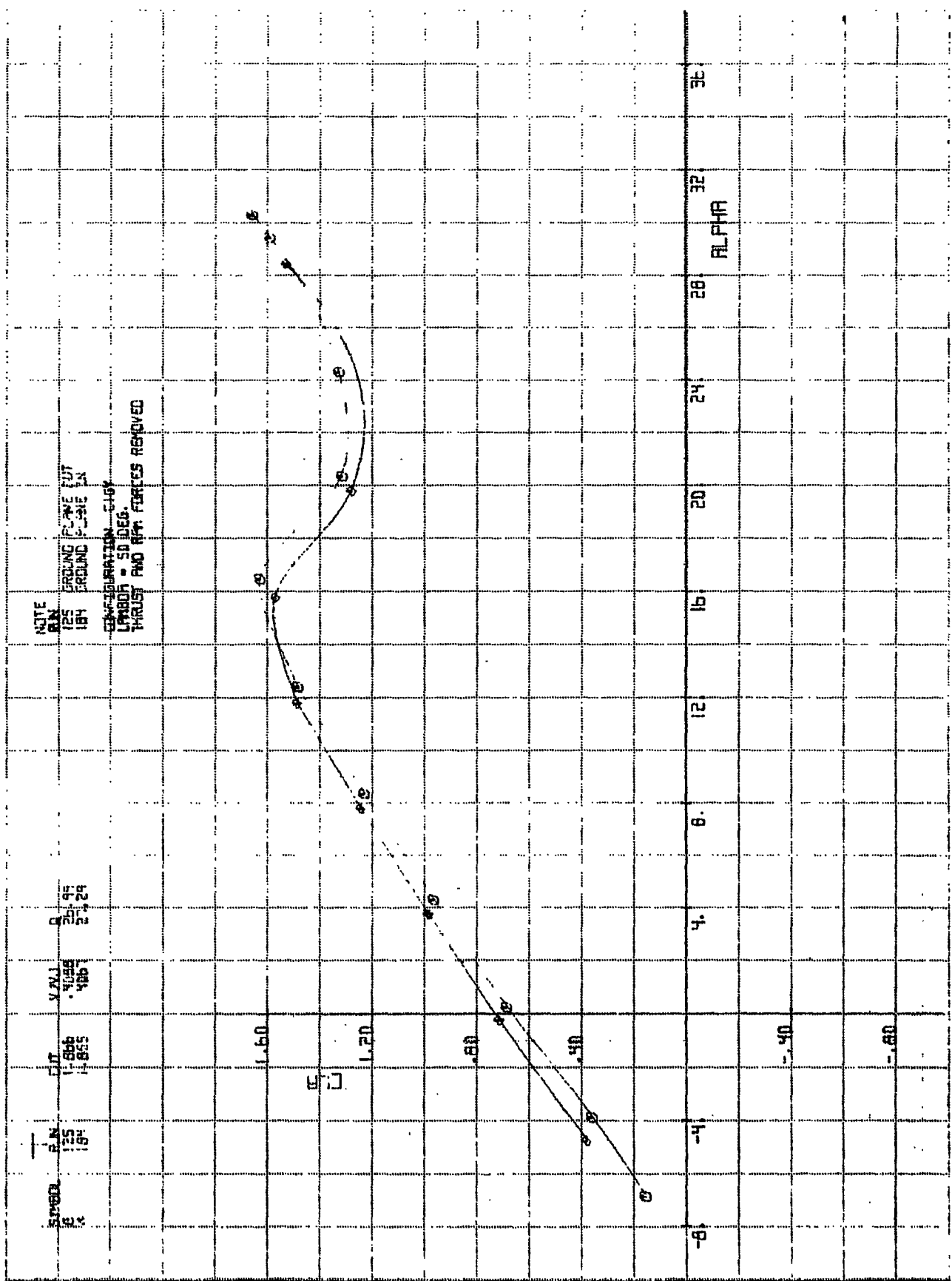
CALC		REVISED	DATE
CHECK			12/10/76
APP			
APP			

CL vs COA
 EFFECT OF GROUND PLANE, LAMBDA = 90
 LUT = 3.1

LEWIS 2541
 RX-1385
 FIG. A. 20
 CONT'D
 PAGE 192



CALC		REVISED	DATE	CLM VS CLA EFFECT OF GROUND PLANE, LAMBDA = 90 CLT = 3.7	LSM-3 ESM
CHECK			12/10/76		HX-1355
APR					FIG. A.20
APR					CONT'D
				THE BOEING COMPANY	PAGE 193



NOTE
CLIN
125
184

GROUND PLANE CUT
125
184

CONFIRMATION - C-119
LAMBDA = 50 DEG.
THRUST AND RPT FORCES REMOVED

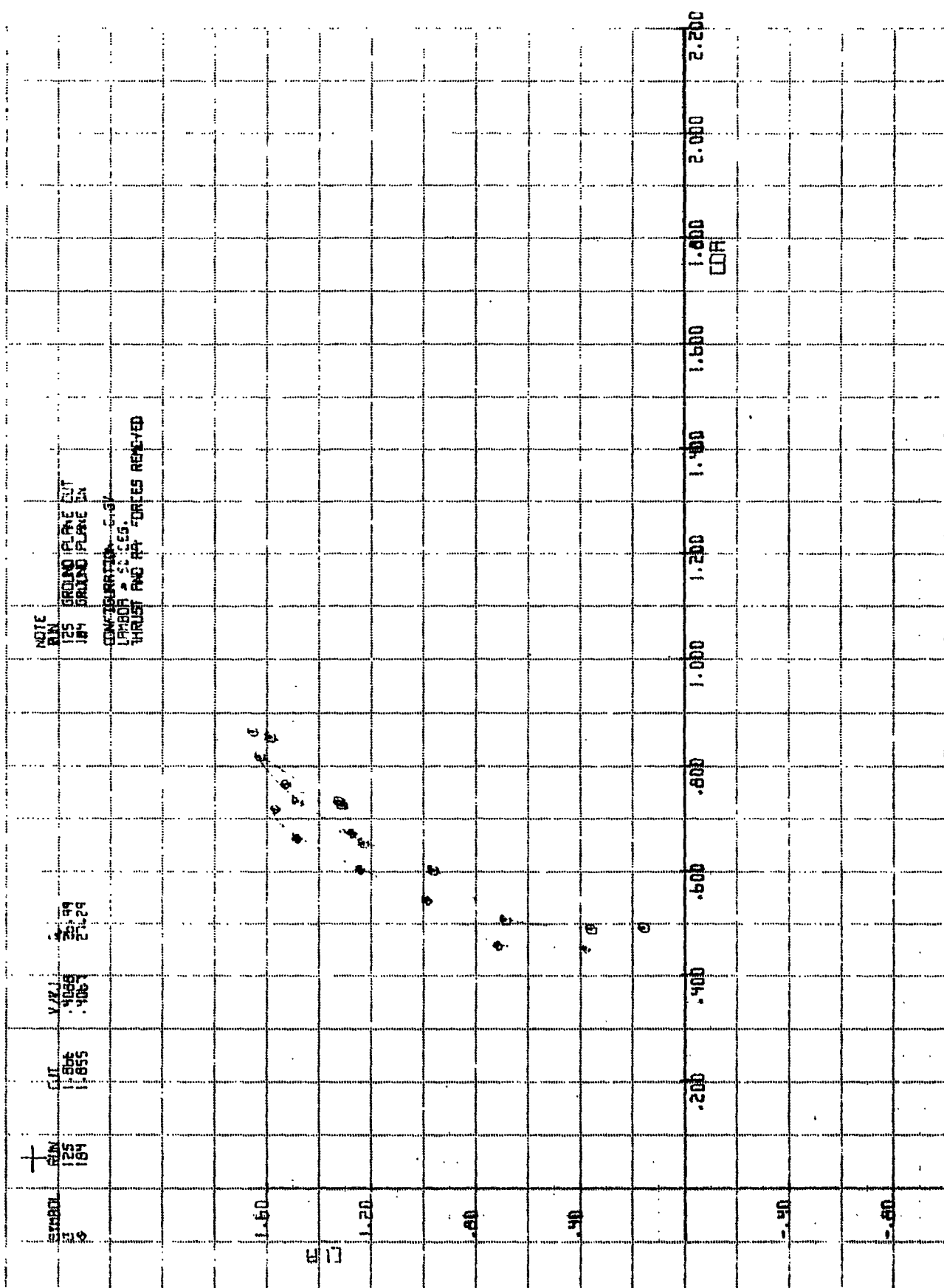
CALC		REVISED	DATE	CL vs ALPHA EFFECT OF GROUND PLANE, LAMBDA = 50 CDT = 1.8	LSMTD 25-11
CHECK			2/10/78		AX-1355
APR					FIG. A.21
APR					PAGE 194
THE BOEING COMPANY					

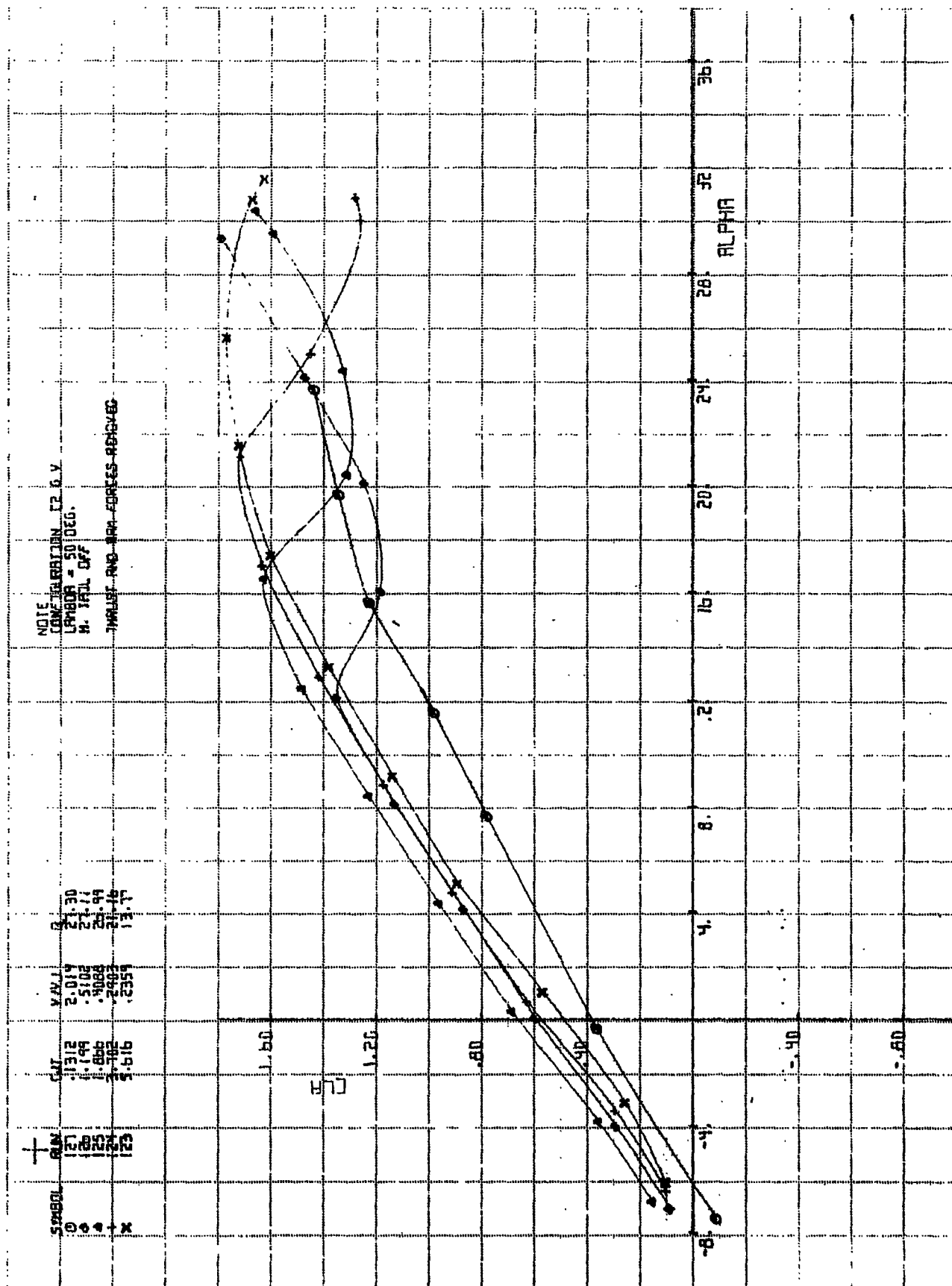
CALC		REVISED	DATE
CHECK			2/10/74
APR			
APR			

CLA VS COA
EFFECT OF GROUND PLANE. LAMBDA = 50°
CIT = 1.8

THE BOEING COMPANY

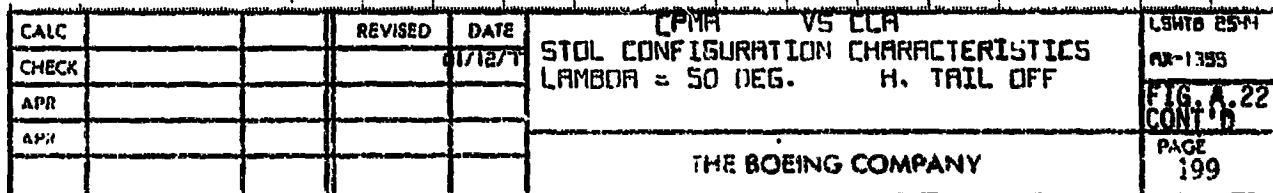
LSMTB 2544
HX-1388
FIG. A. 21
CONT'D
PAGE 195



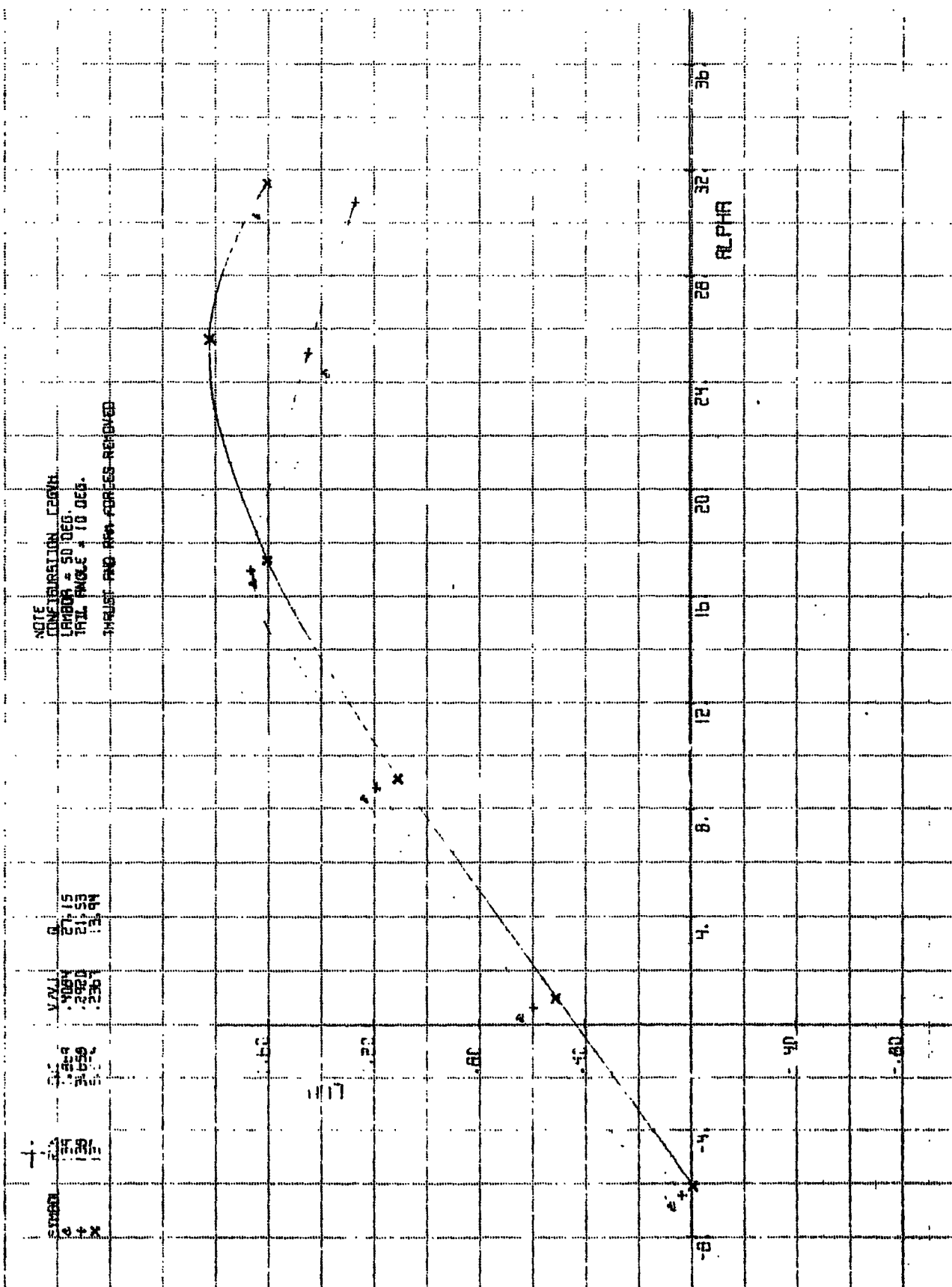


CALC		REVISED	DATE	CL vs ALPHA	SHED 2544
CHECK			01/12/71	STOL CONFIGURATION CHARACTERISTICS	AX-1355
APR				LAMBDA = 50 DEG. H. TAIL OFF	FIG.A.22
APR					PAGE 197
THE BOEING COMPANY					

NOTE
CONCENTRATION 12 G.V.
LAWAER - 50 DEG.
H. TAIL OFF

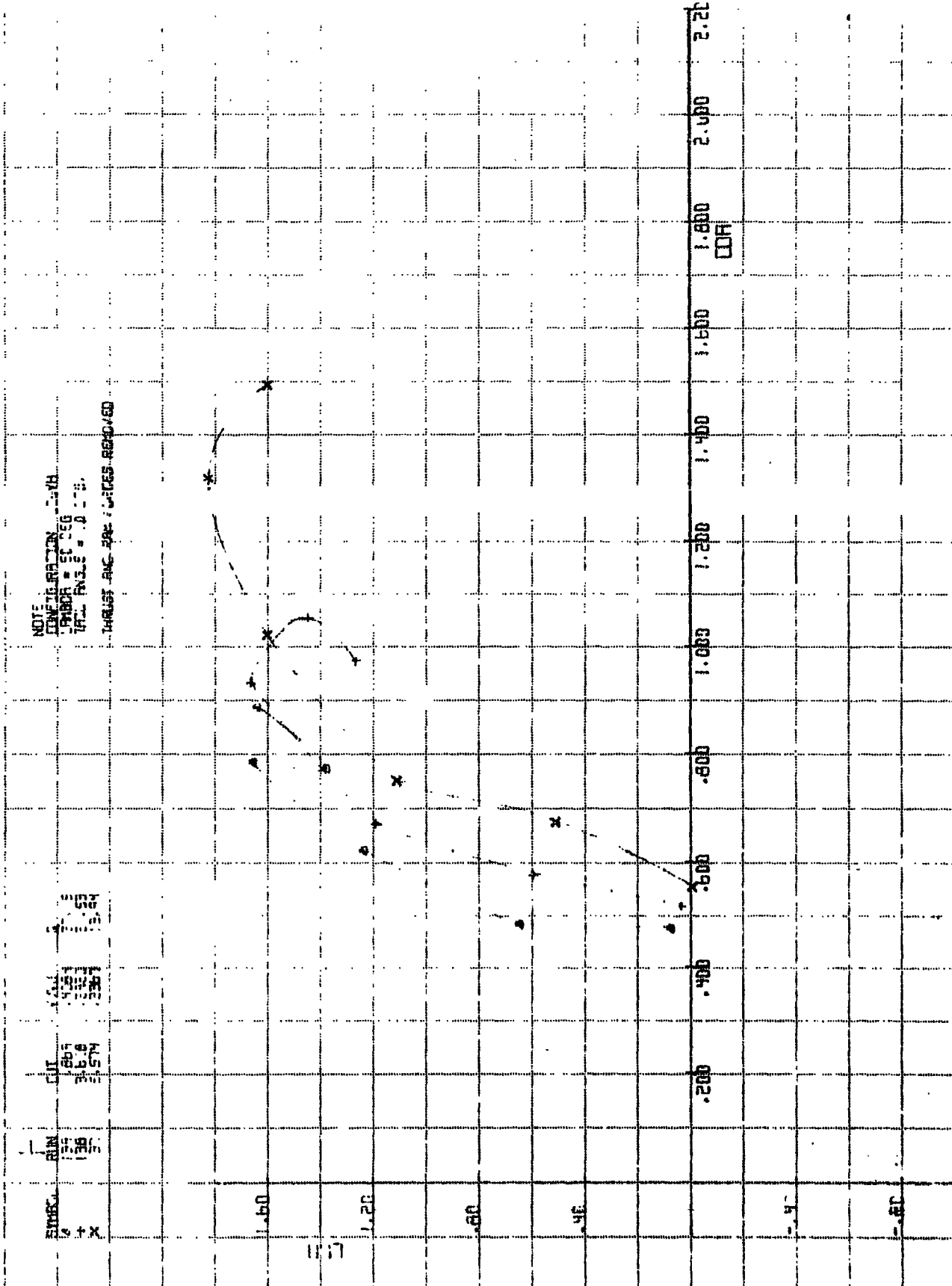


NOTE
 CONFIGURATION L204H
 LAMBDA = 50 DEG
 TAIL ANGLE = 10 DEG
 THRUST AND AIR FORCES REMOVED



CALC		REVISED	DATE	CL vs ALPHA	LEWIS ES-11
CHECK			2/10/75	STOL CONFIGURATION CHARACTERISTICS	FX-1385
APR				LAMBDA = 50 DEG, TAIL ANGLE = 10 DEG	FIG.A.23
APR					PAGE 200
THE BOEING COMPANY					

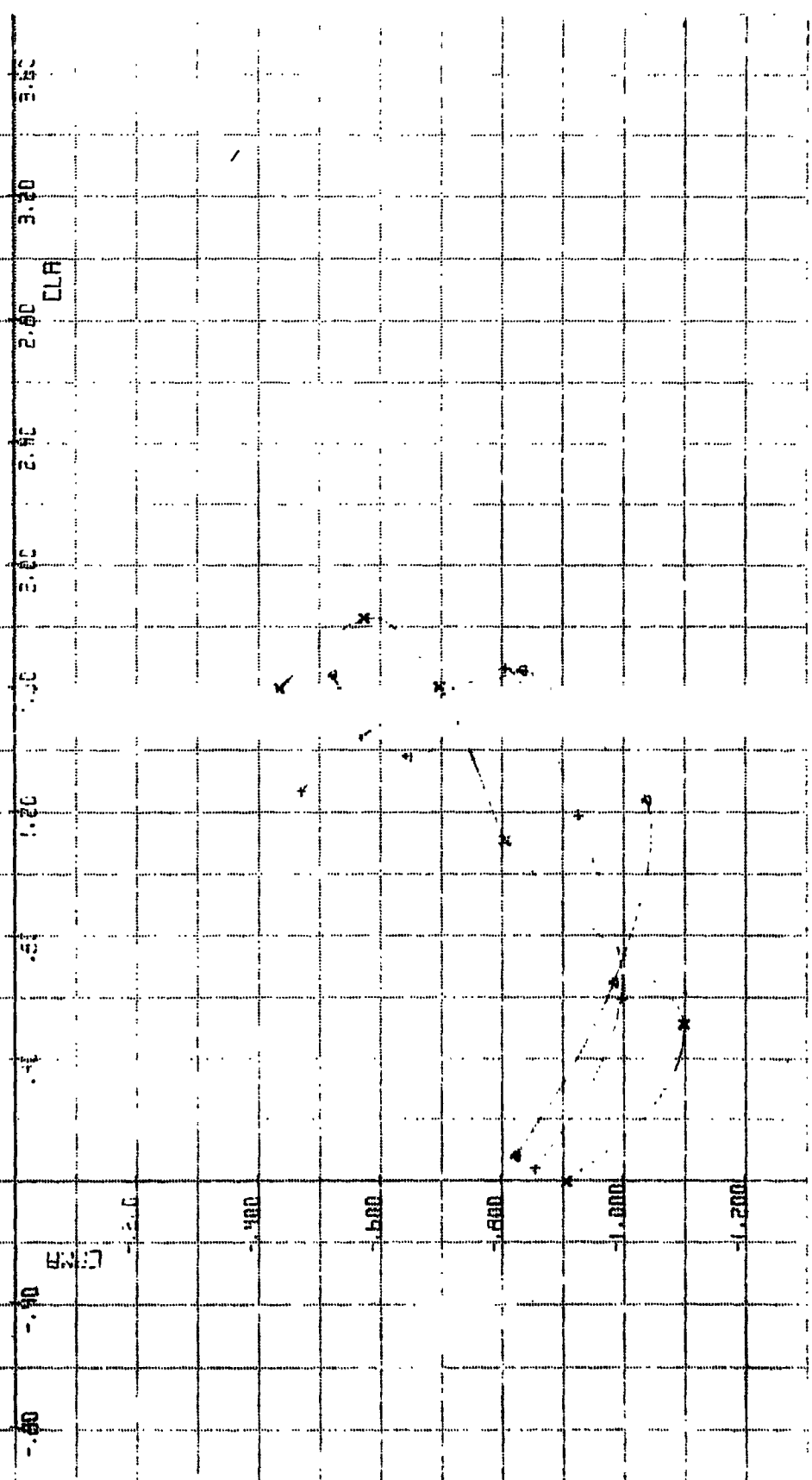
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CALC				REVISED	DATE	LIFT V.C. CDF STILL CONTINUATION CHARACTERISTICS THROU = 70 DEG. THIS ANGLE = 10 DEG			LSWTR 25-4 MX 1325 FIG. A.23 CONT'D PAGE 201																
CHECK				12/10/76																					
APR																									
QUR																									



NOTE: AIRBORNE TESTS
 WERE MADE AT 5000 FT.
 ALTITUDE AND 100 MPH
 WIND SPEED.

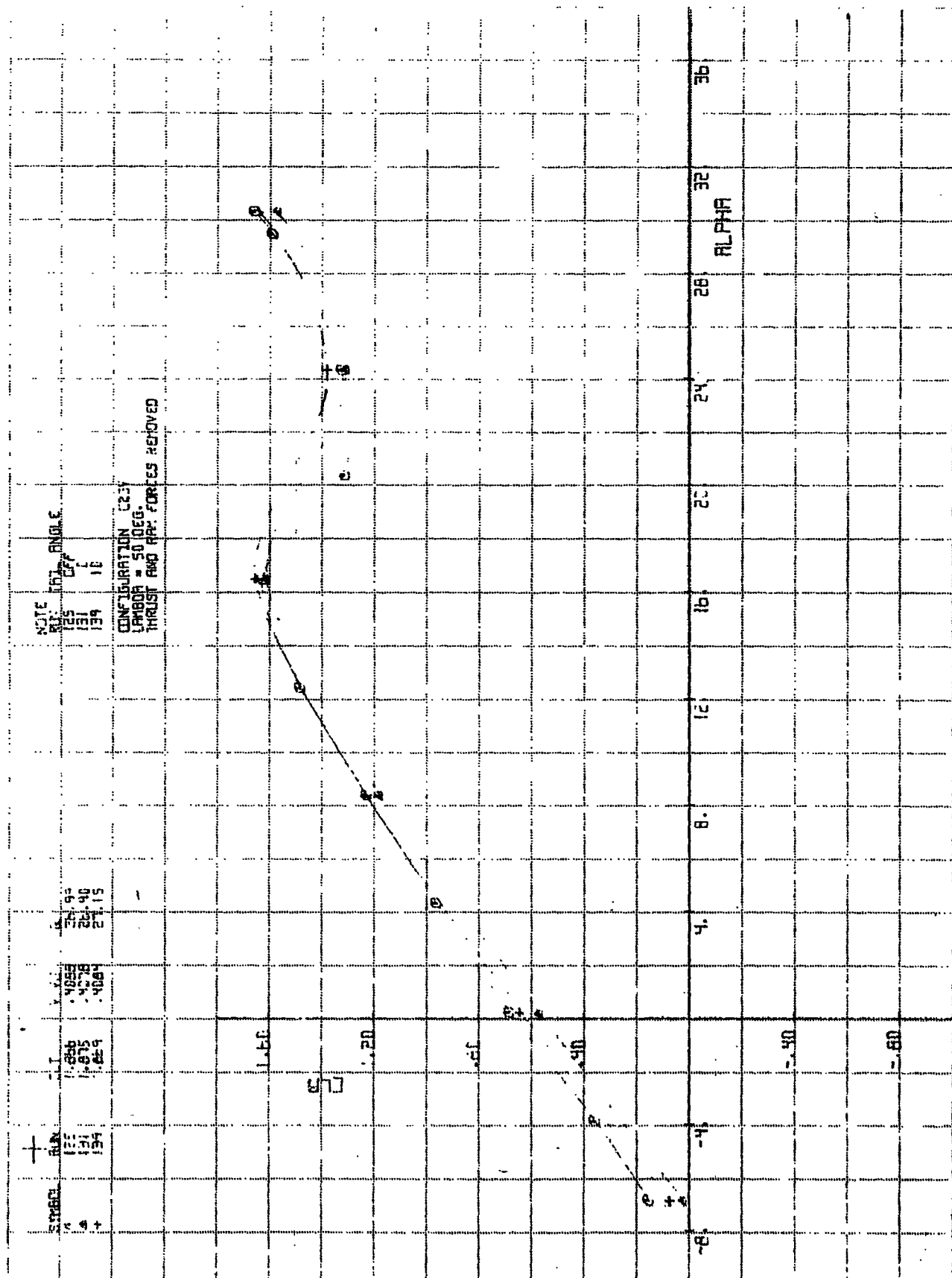
1.000
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 10.000

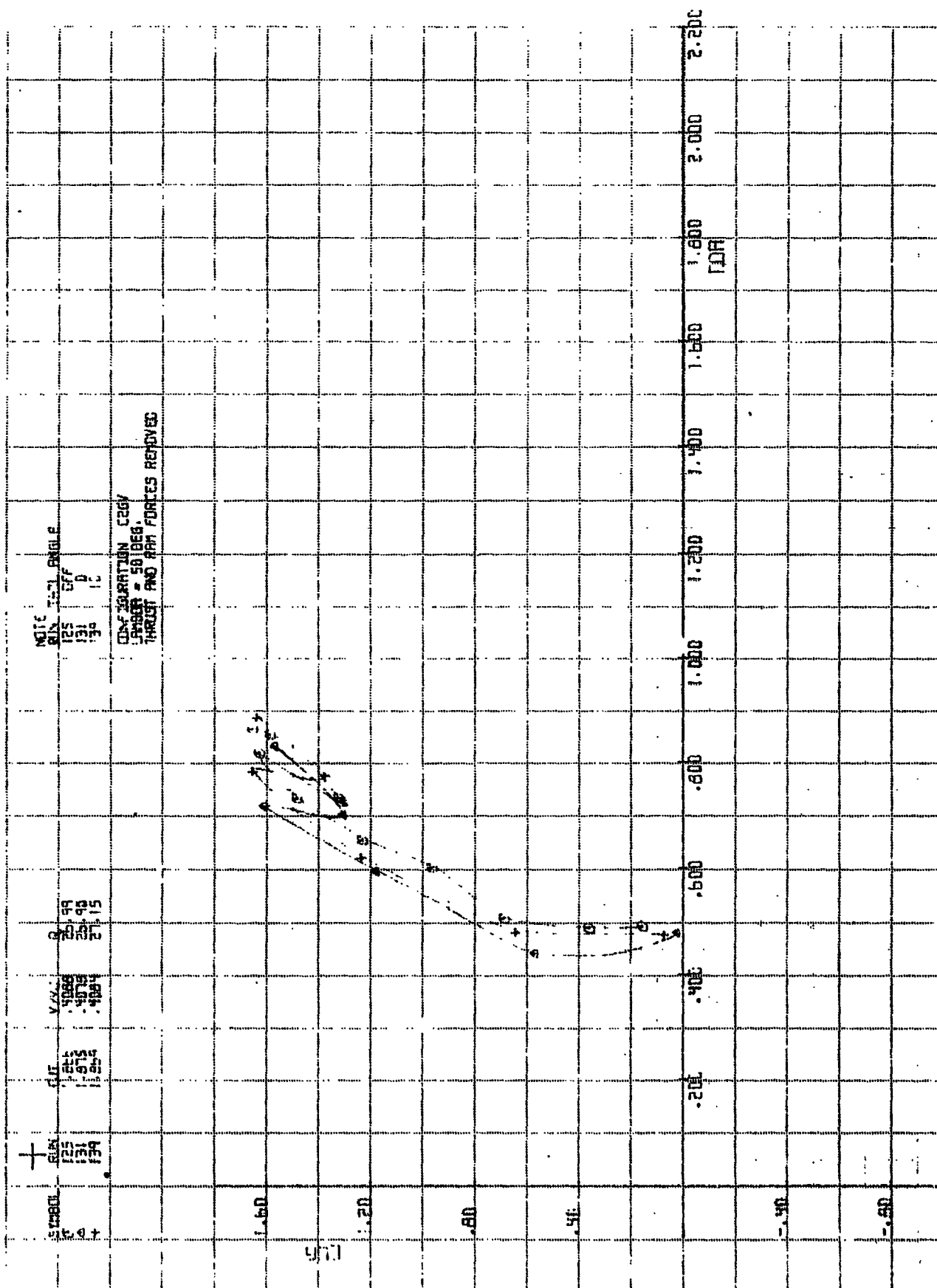


DATE	REVISED	DATE	BY	VS CLM	CLM TO CLM
12/10/77				CLM TO CLM	CLM TO CLM
APR				CLM TO CLM	CLM TO CLM
APR				CLM TO CLM	CLM TO CLM
THE BOEING COMPANY					PAGE 202 CONT'D

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CALC		REVISED	DATE	<p>VS ALPHA EFFECT OF HORIZONTAL TAIL LAMBDA = 50 DEG, C.T. = 1.8</p>	LSMTB (254)
CHECK			12/10/77		AX-1855
APR					FIG. A.24
APR					PAGE 203
				THE BOEING COMPANY	

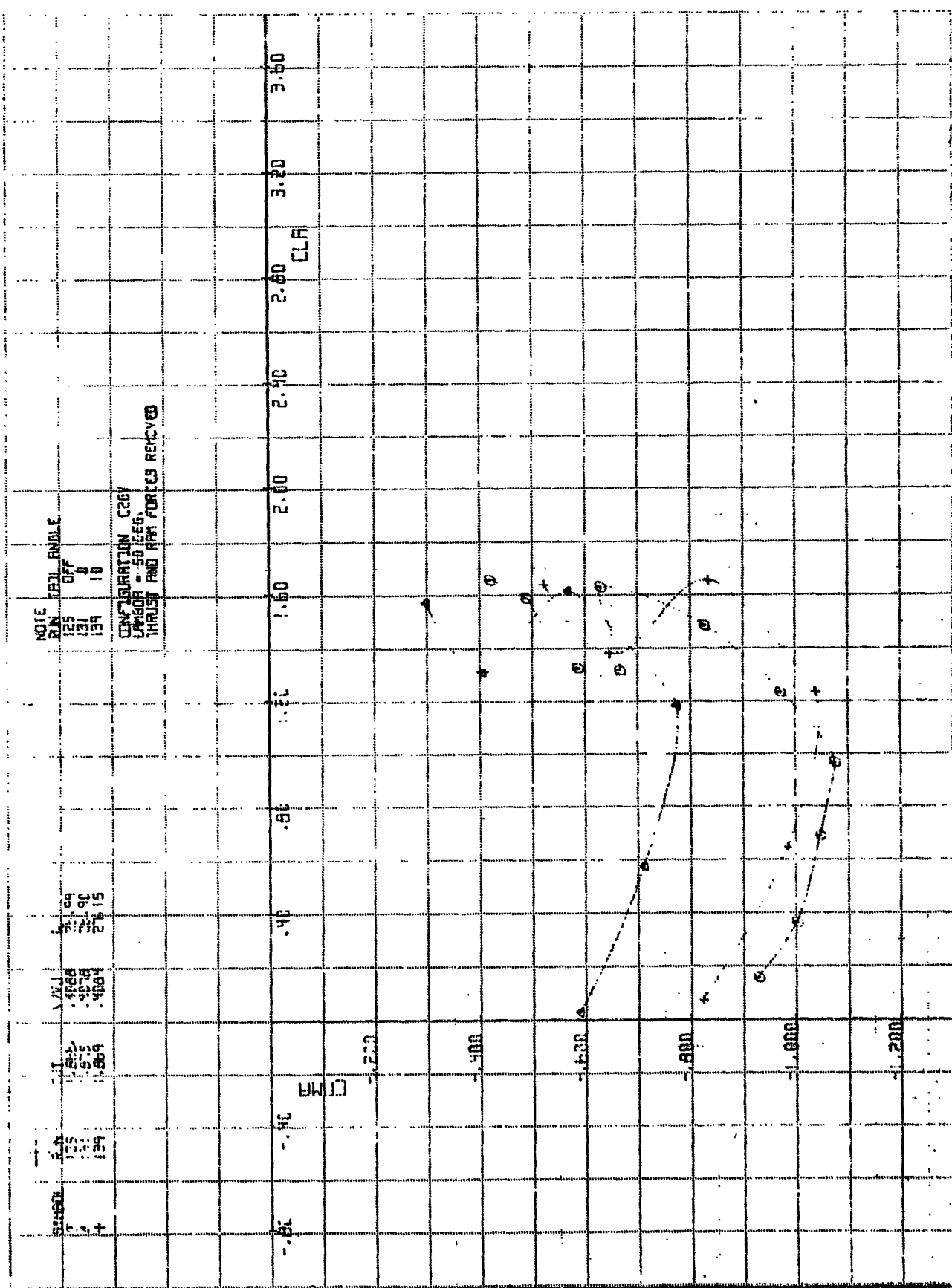


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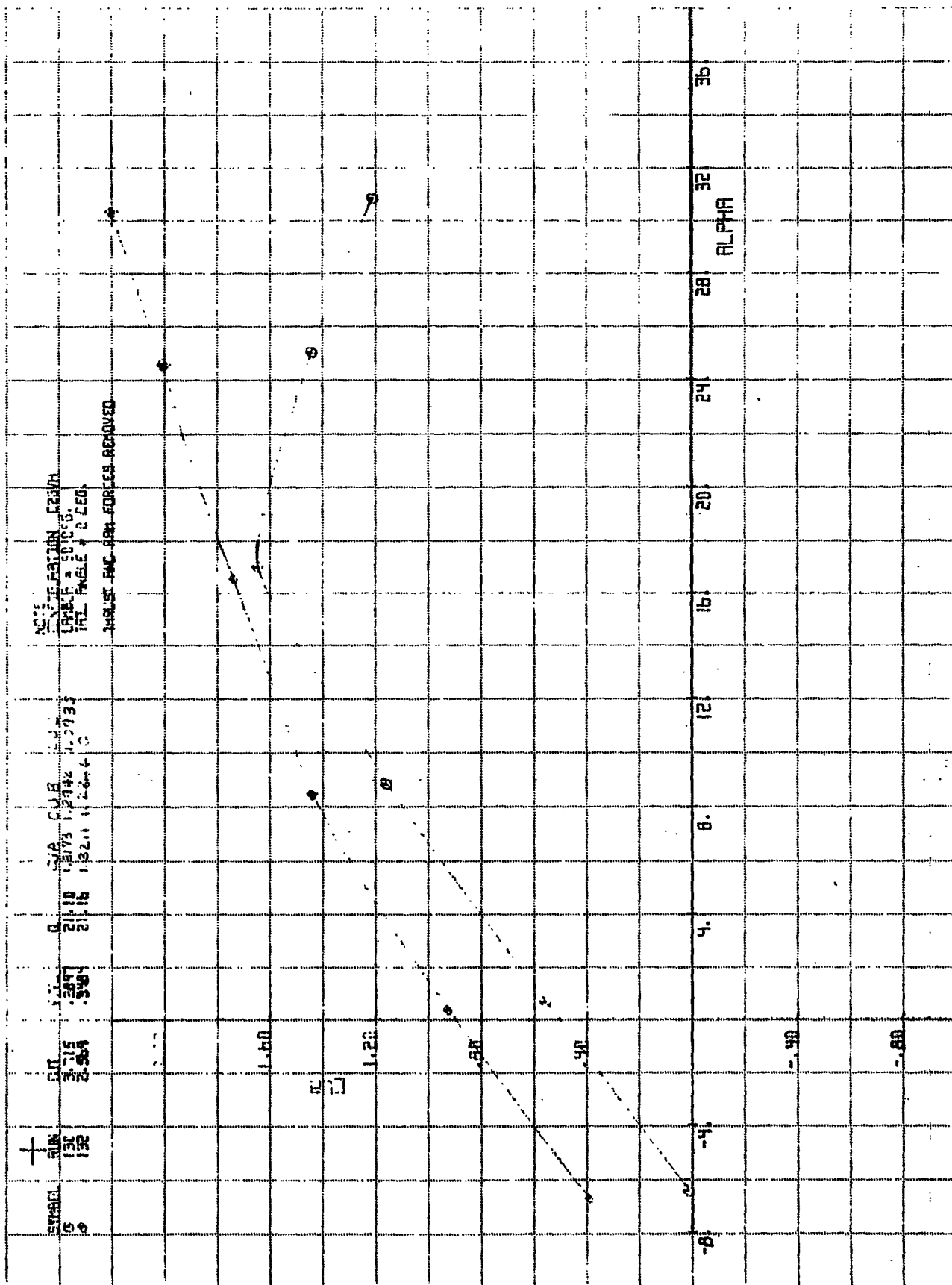
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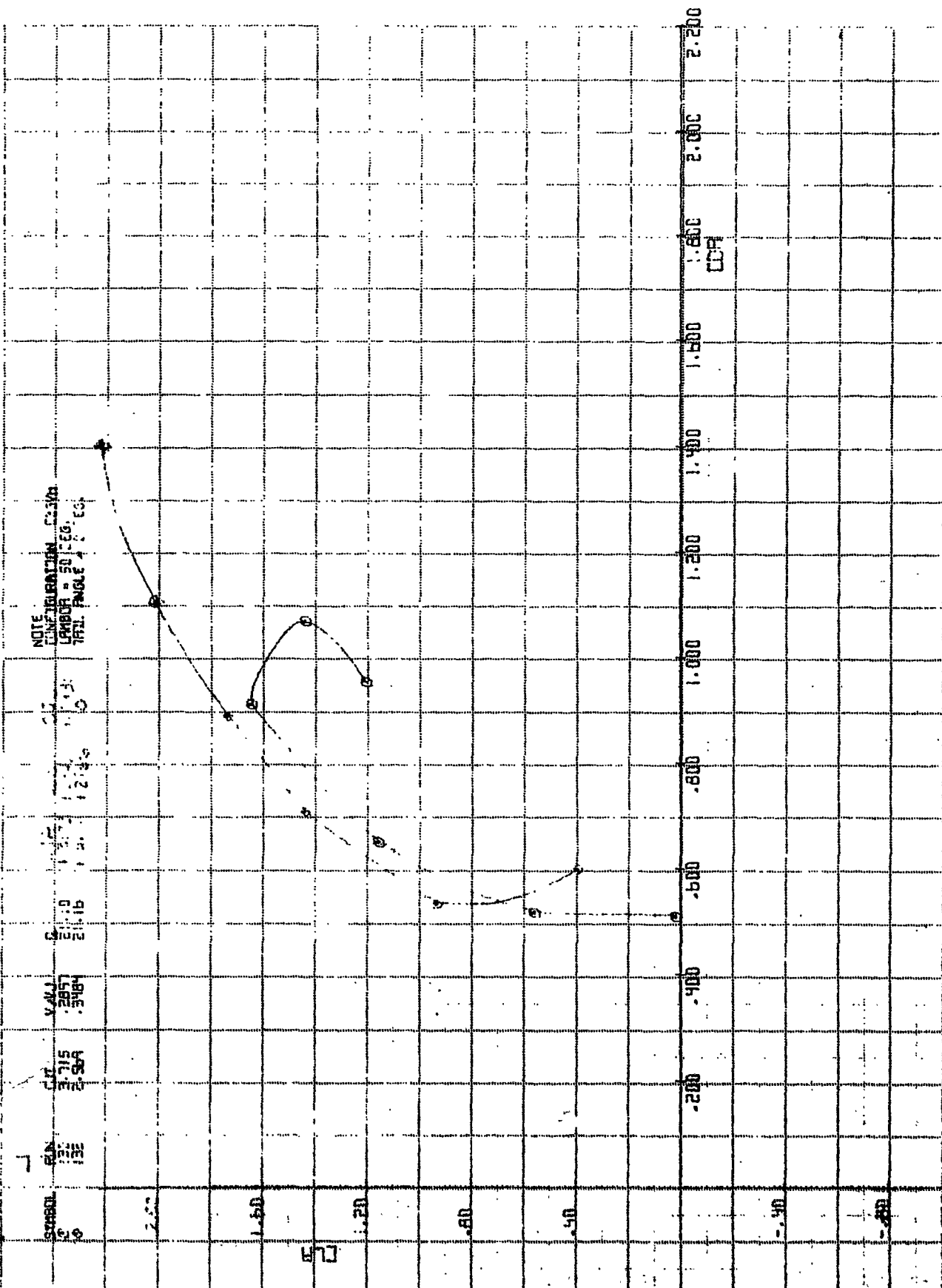
FIG 14
14.



CALC		REVISED	DATE	CPMA VS CLM	LSMTB 2544
CHECK			2/10/76	EFFECT OF HORIZONTAL TAIL	AX-1335
APR				LAMBDA = 50 DEG, CLT = 1.8	FIG. A.24
APR					CONT'D
				THE BOEING COMPANY	PAGE 203



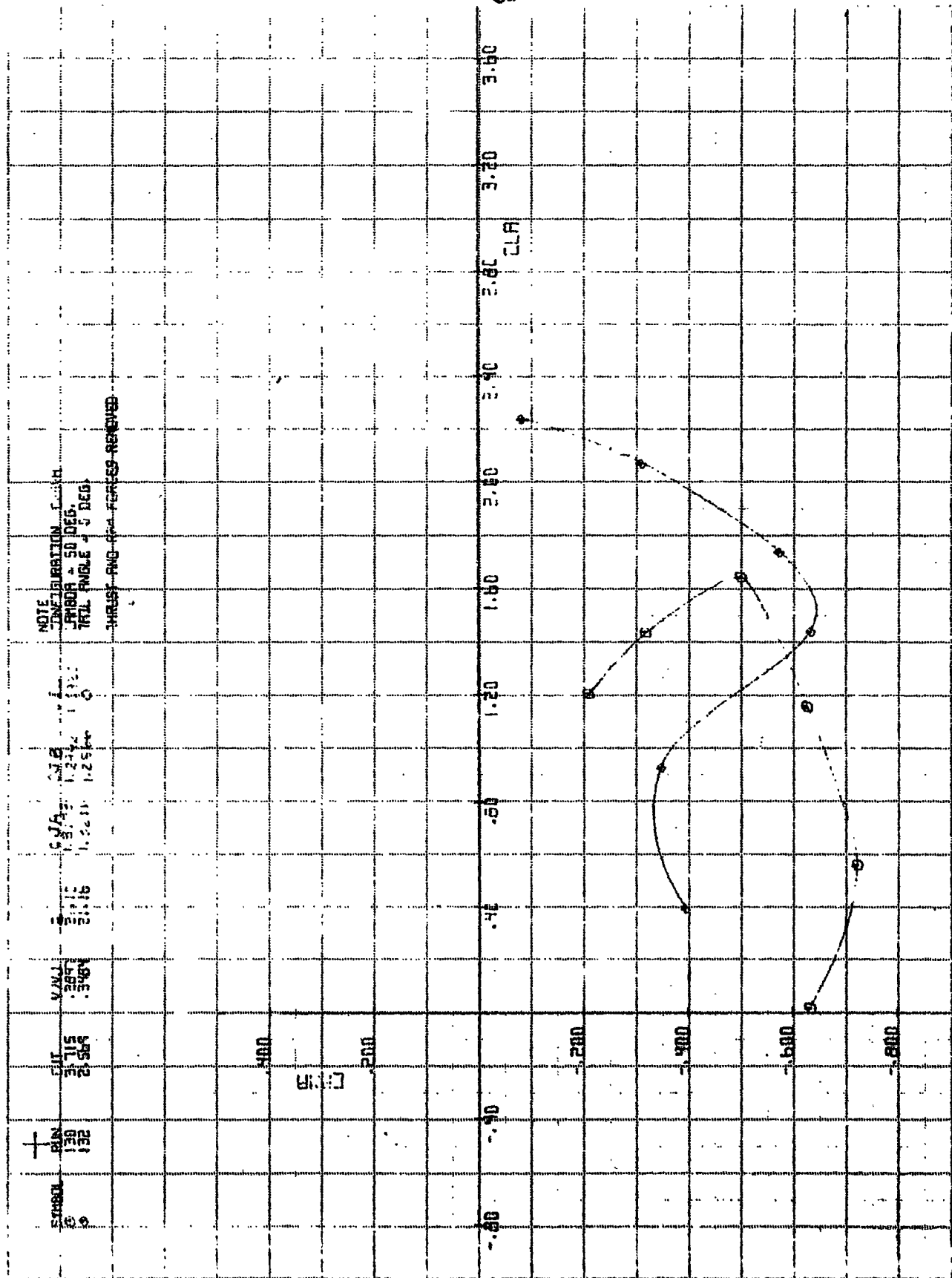




CALC		REVISED	DATE	CLH VS LDR	LEAD 8544
CHECK			2/10/72	EFFECT OF NOSE FIN THRUST VARIATION	AX-1555
APR				THRUST = 50 DEG.	FIG. A-26
APR					CONT. 6
					PAGE
					210

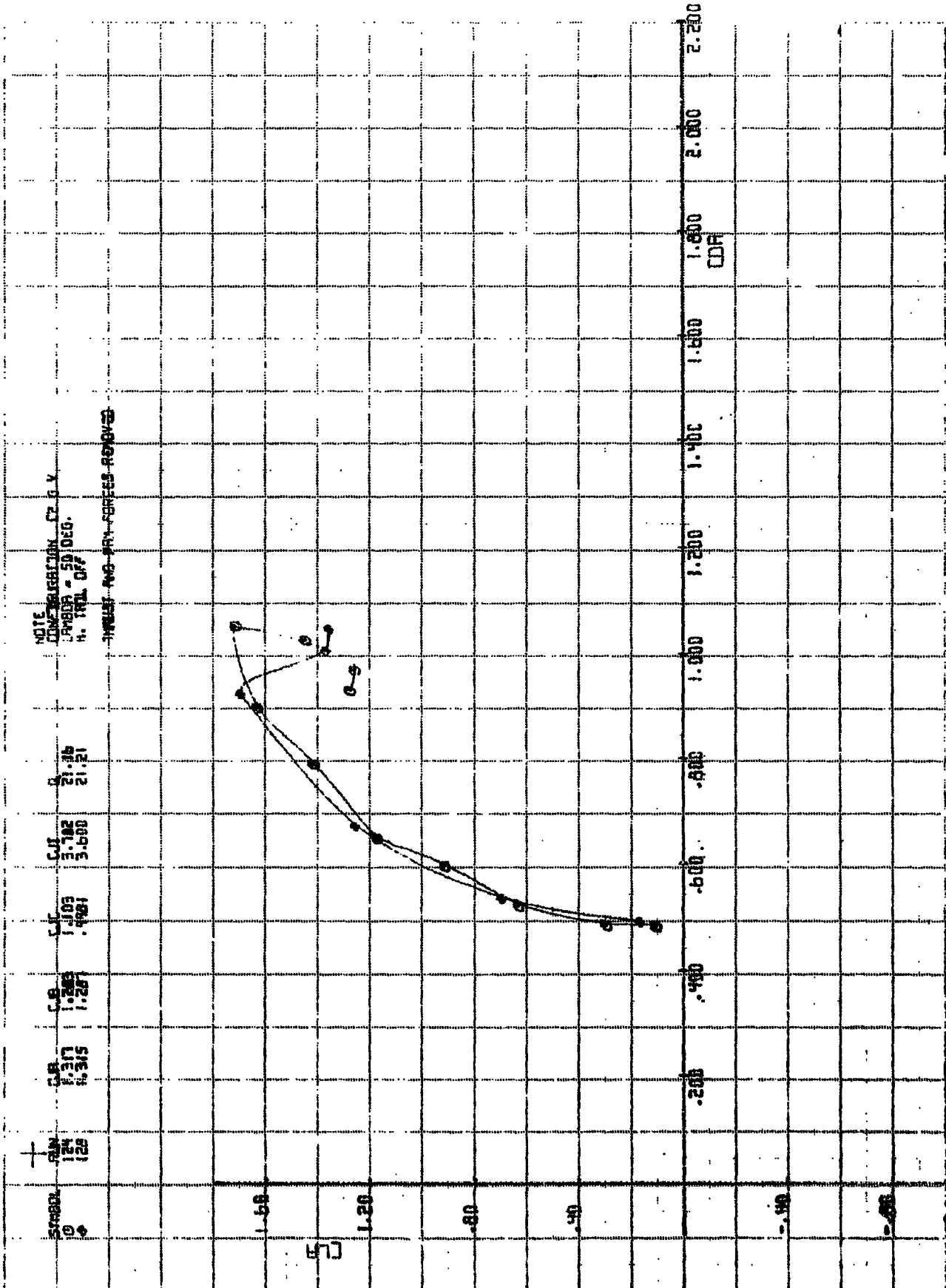
THE BOEING COMPANY

ORIGINAL PAGE 12
OF POOR QUALITY

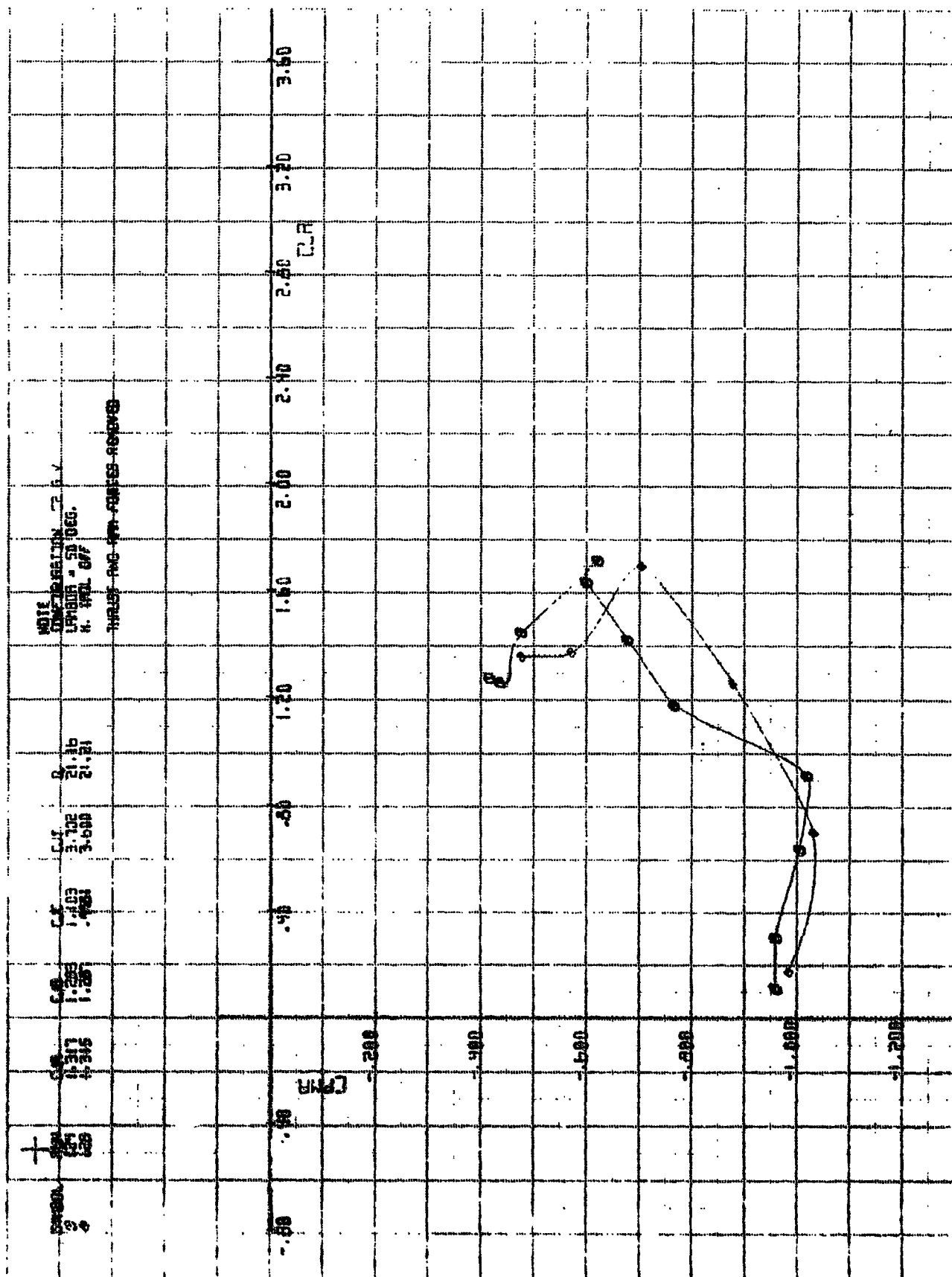


NOTE
CONFIDENCE LIMIT
LAMBDA = 50 DEG.
TRAIL ANGLE = 5 DEG.
THRUST AND AREA FACTORS REMOVED

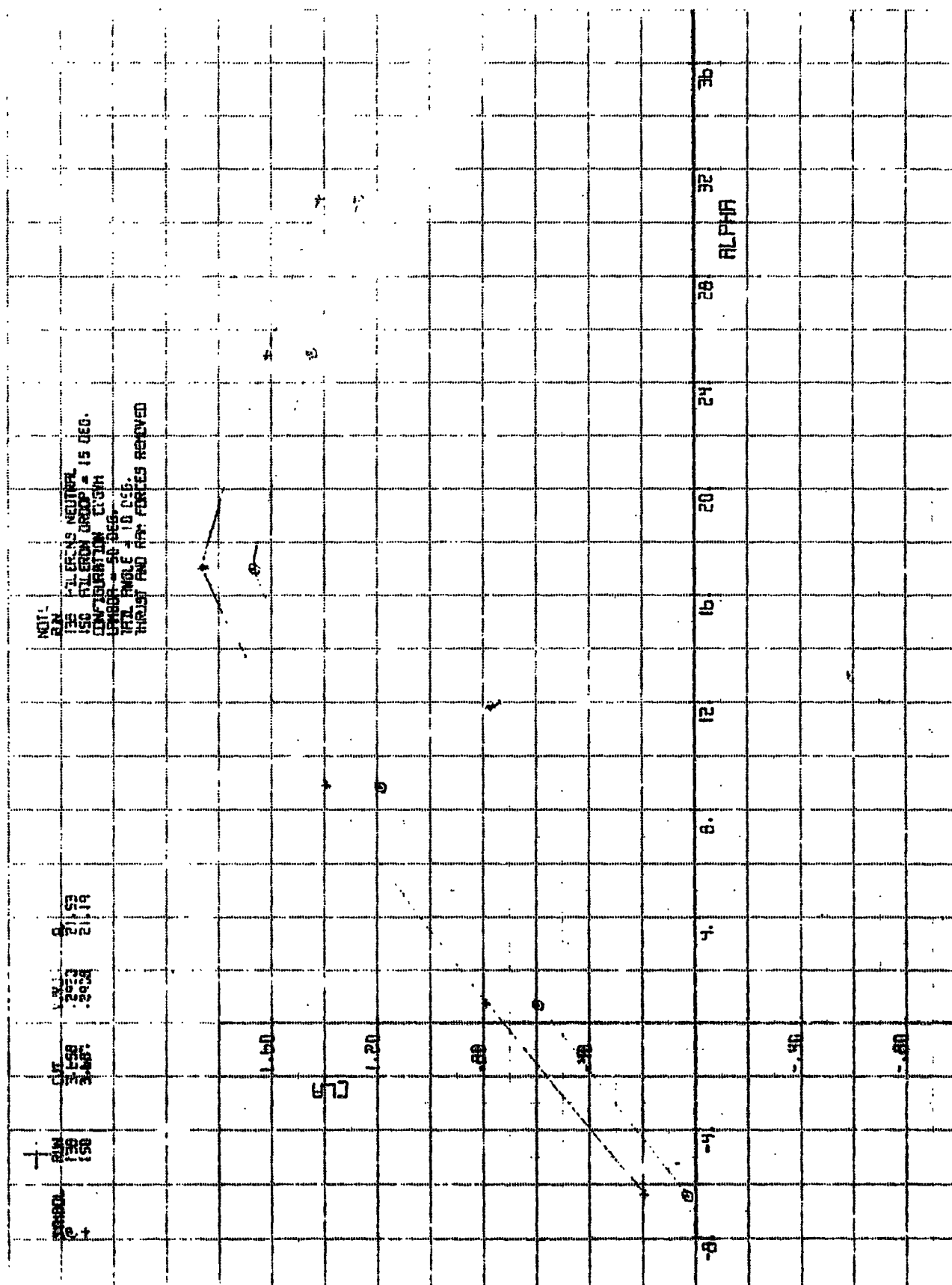
ALC		REVISED	DATE	EFFECT OF NOZZLE EXIT THRUST VARIATION LAMBDA = 50 DEG. THE BOEING COMPANY	LAMDA 50 DEG 11-1-128 FIG. A.26 CONT'D PAGE 211
CHECK			2/10/78		
APR					
APR					



CL	VS	CD
EFFECT OF NOSE FAN THRUST VARIATION		
LAMBDA = 90 DEG. H. TAIL OFF		
THE BOEING COMPANY	213	213



CALC		REVISED	DATE	CPMR VS CLR EFFECT OF NOSE FAN THRUST VARIATION LAMBDA = 50 DEG. H. TAIL OFF	LEMBO 254 PM-1355 FIG. A. 27 CONT'D PAGE 214
CHKD					
APR					
APR					

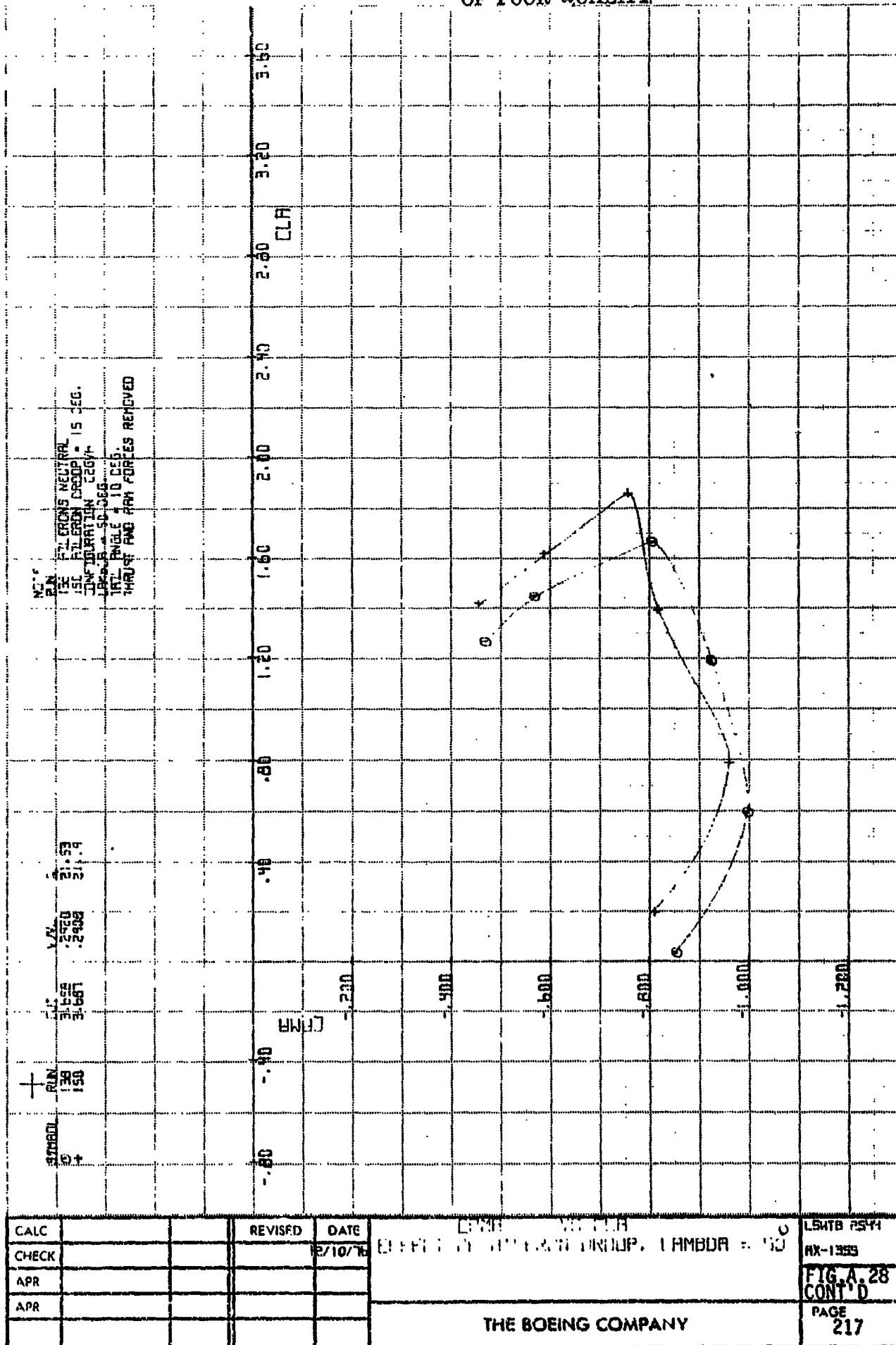
[illegible]

CALC		REVISED	DATE	CLC	VS. AIRCRAFT	LIMITED BY
CHECK			10/7/76	EFFECT OF AILERON DROOP, LAMBDA = 50°		NO 13/76
APP						FIG.A.28
APP				THE BOEING COMPANY		PAGE 215

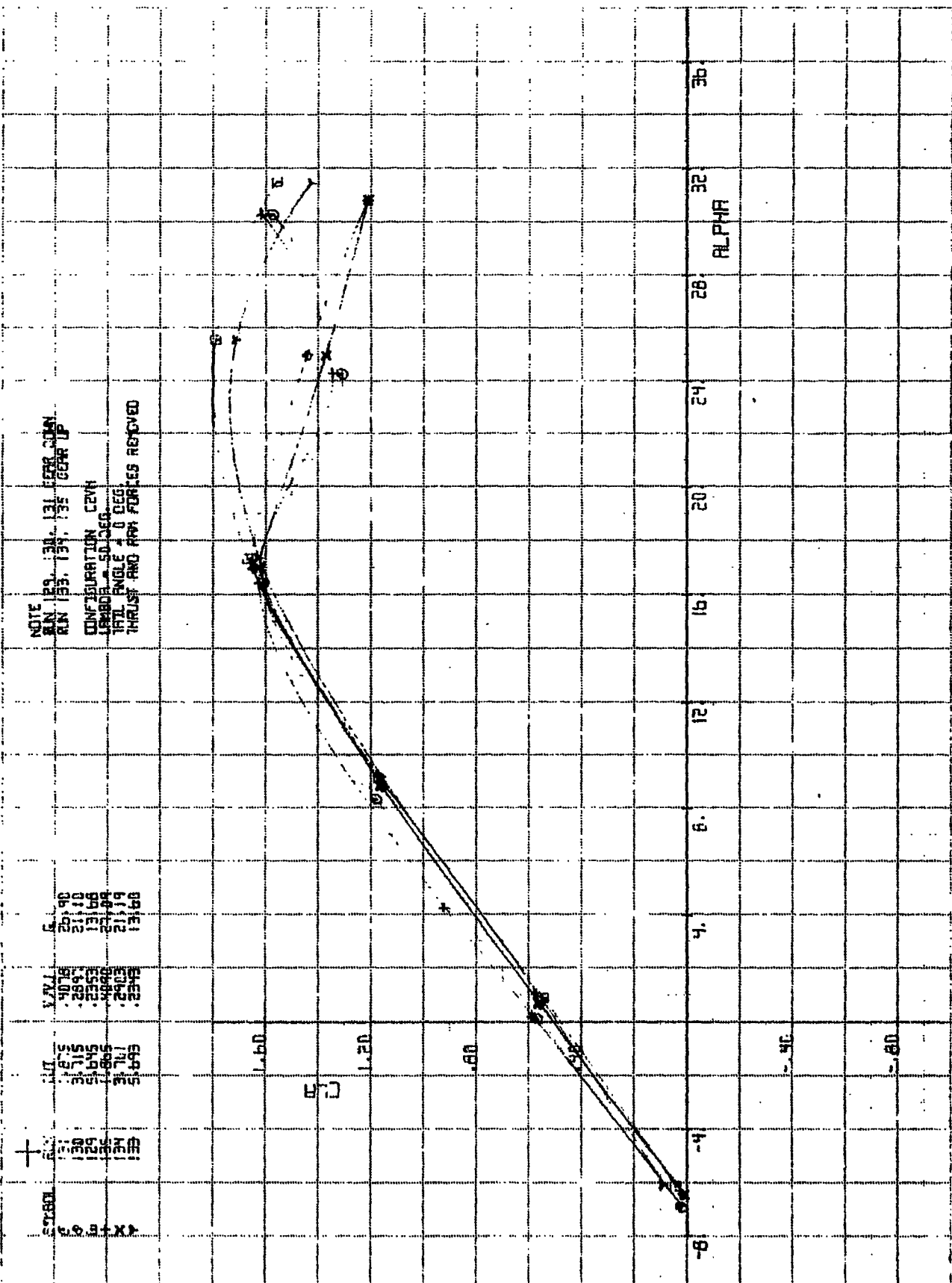
1-2-1

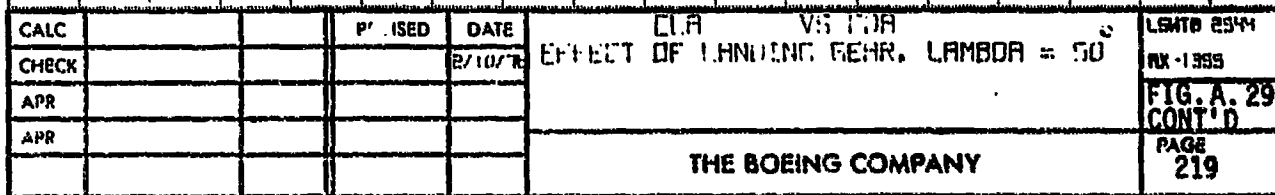
1-33

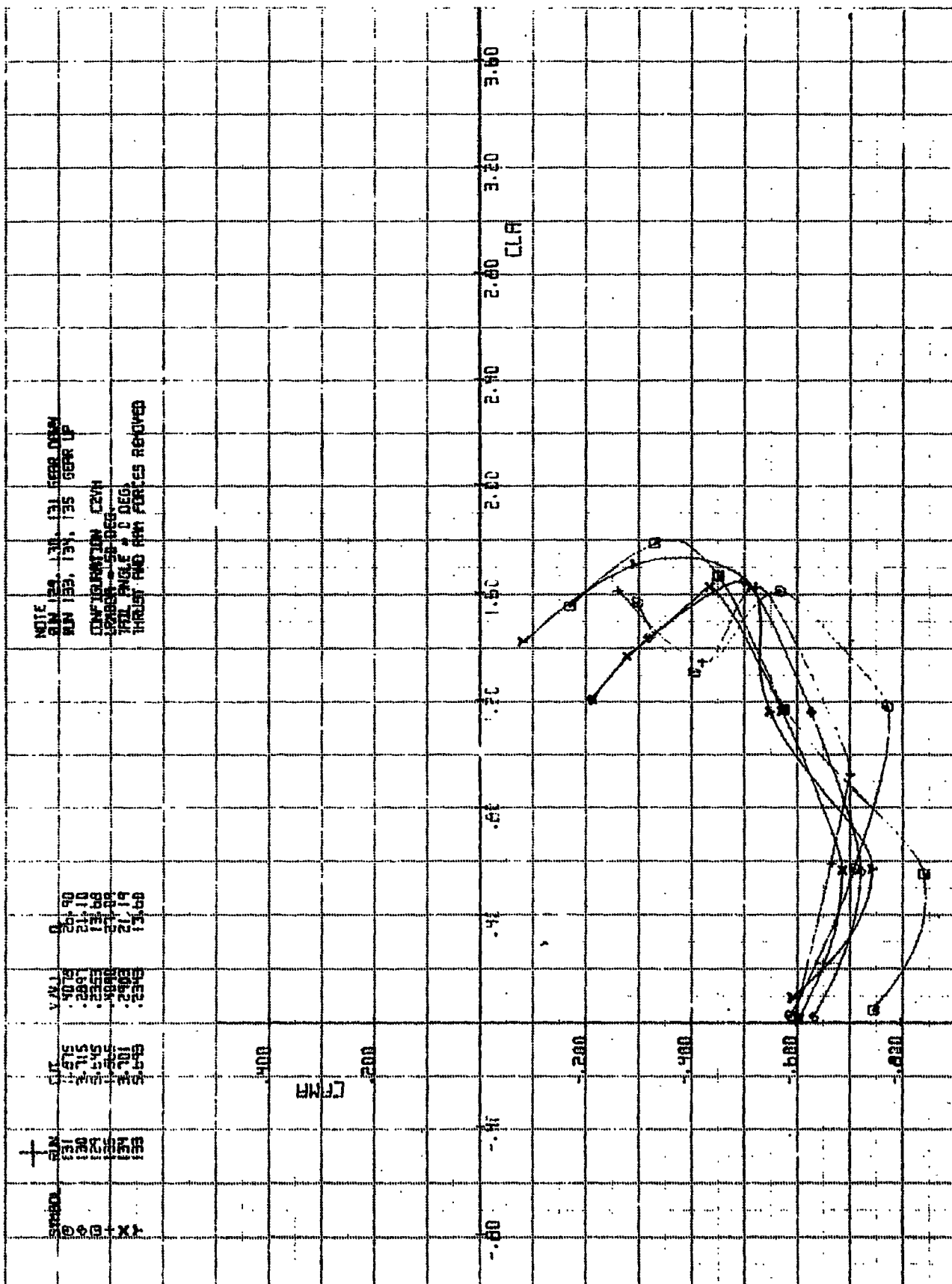
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OF POOR QUALITY



CALC			REVISED	DATE	LIMIT VECTOR EFFECT OF FILTRON DROOP, LAMBDA = 50	LSMTB 2544
CHECK				2/10/76		AX-1333
APR						FIG. A.28
APR						CONT'D
						PAGE 217
THE BOEING COMPANY						





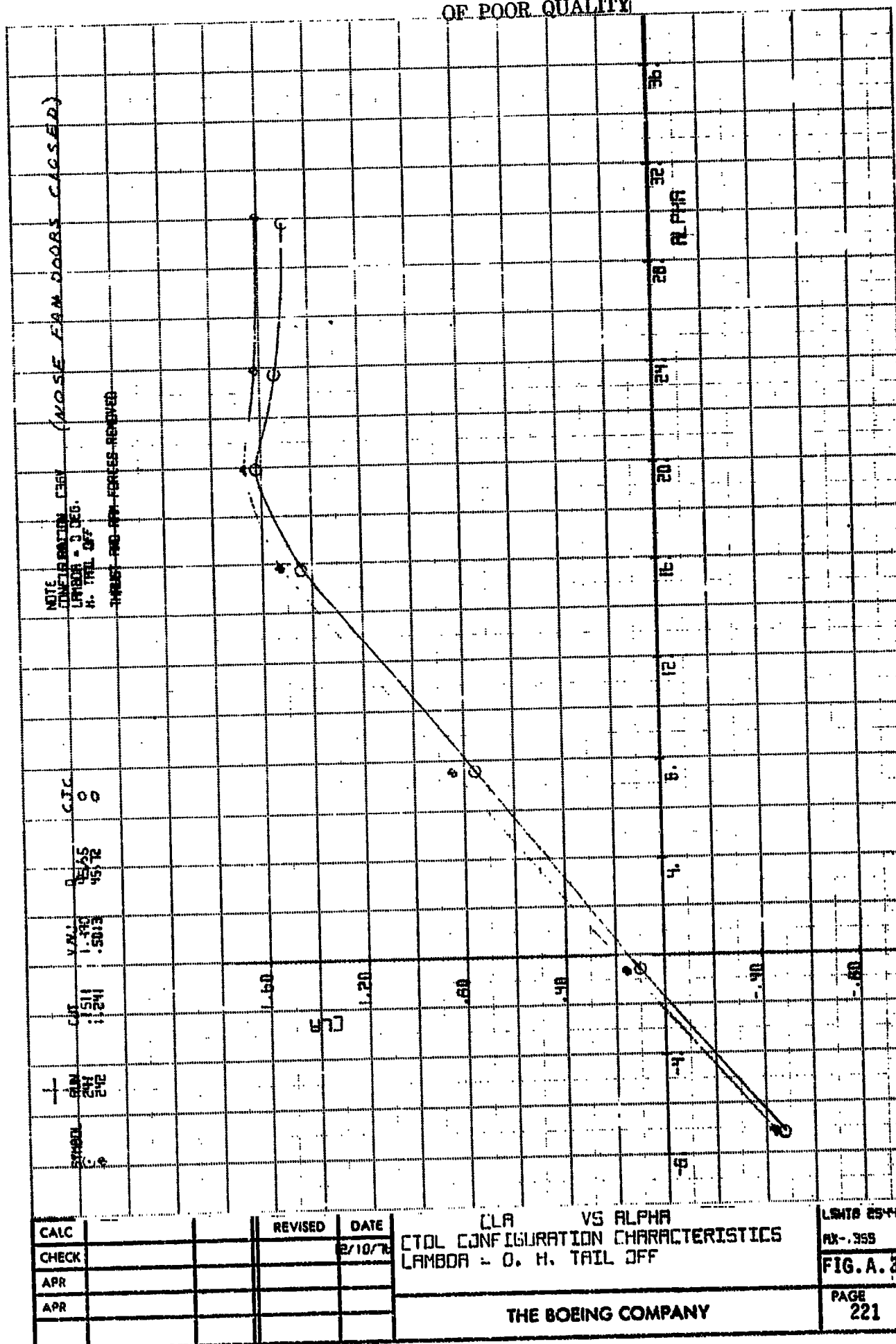


CALC		REVISED	DATE
CHECK			2/10/78
APR			
APR			

CLMA VS CLM
EFFECT OF LANDING GEAR. LAMBOA = 50

LSMTB 2344
RX 1335
FIG. A.29
CONT'D
PAGE
220

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OF POOR QUALITY

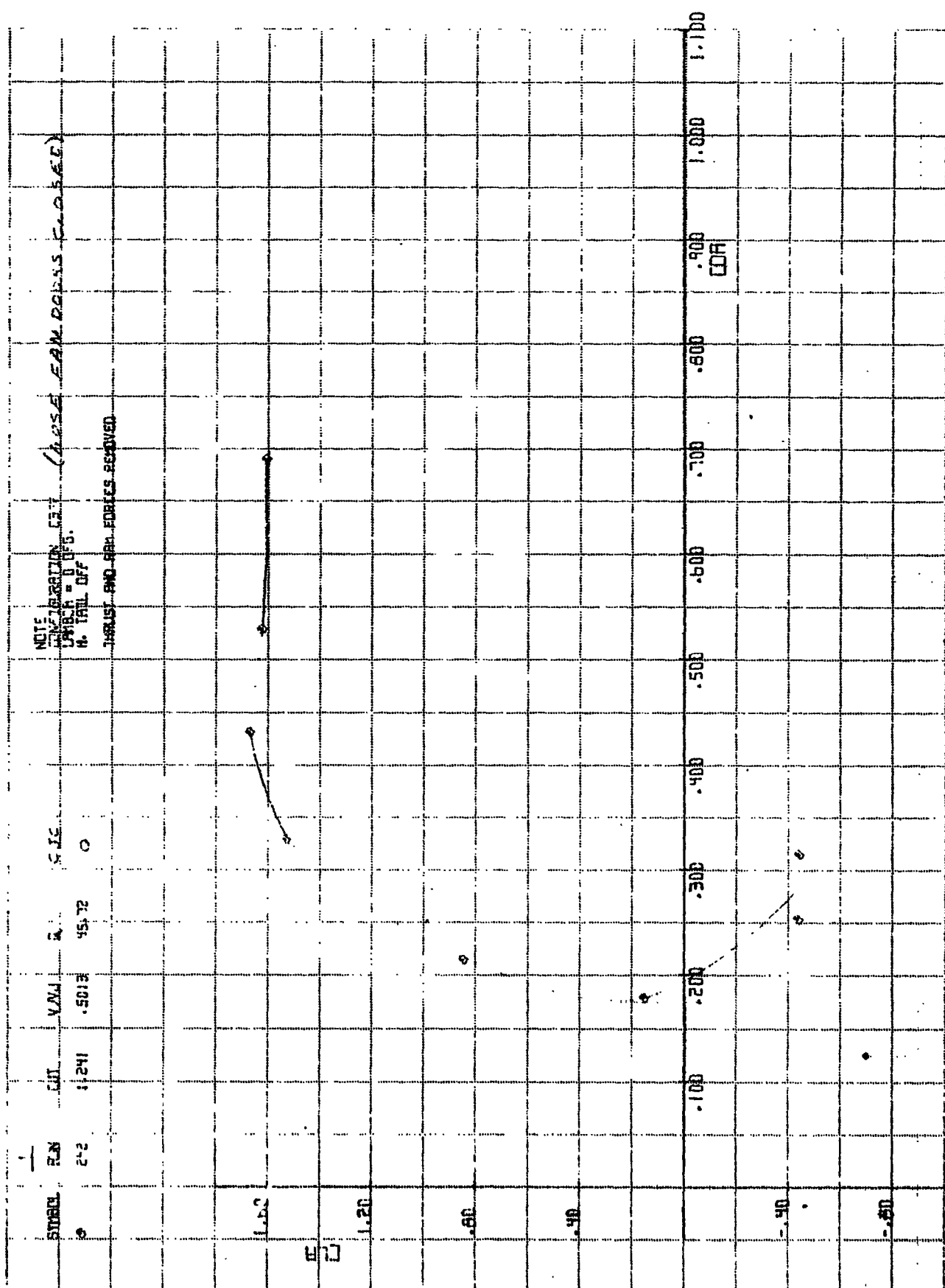


CALC		REVISED	DATE
CHECK			2/10/76
APR			
APR			

CL vs ALPHA
CTOL CONFIGURATION CHARACTERISTICS
LAMBDA = 0. H. TAIL OFF

THE BOEING COMPANY

LSMTB 2544
AX-355
FIG. A. 30
PAGE
221



CLC		REVISED	DATE	CLA VS CDR	LSH 2541
CHECK			12/10/76	CTOL CONFIGURATION CHARACTERISTICS	AX-1355
APR				LAMBDA = 0. H. TAIL OFF	FIG. A 30
APR					CONT'D
					PAGE 222

THE BOEING COMPANY

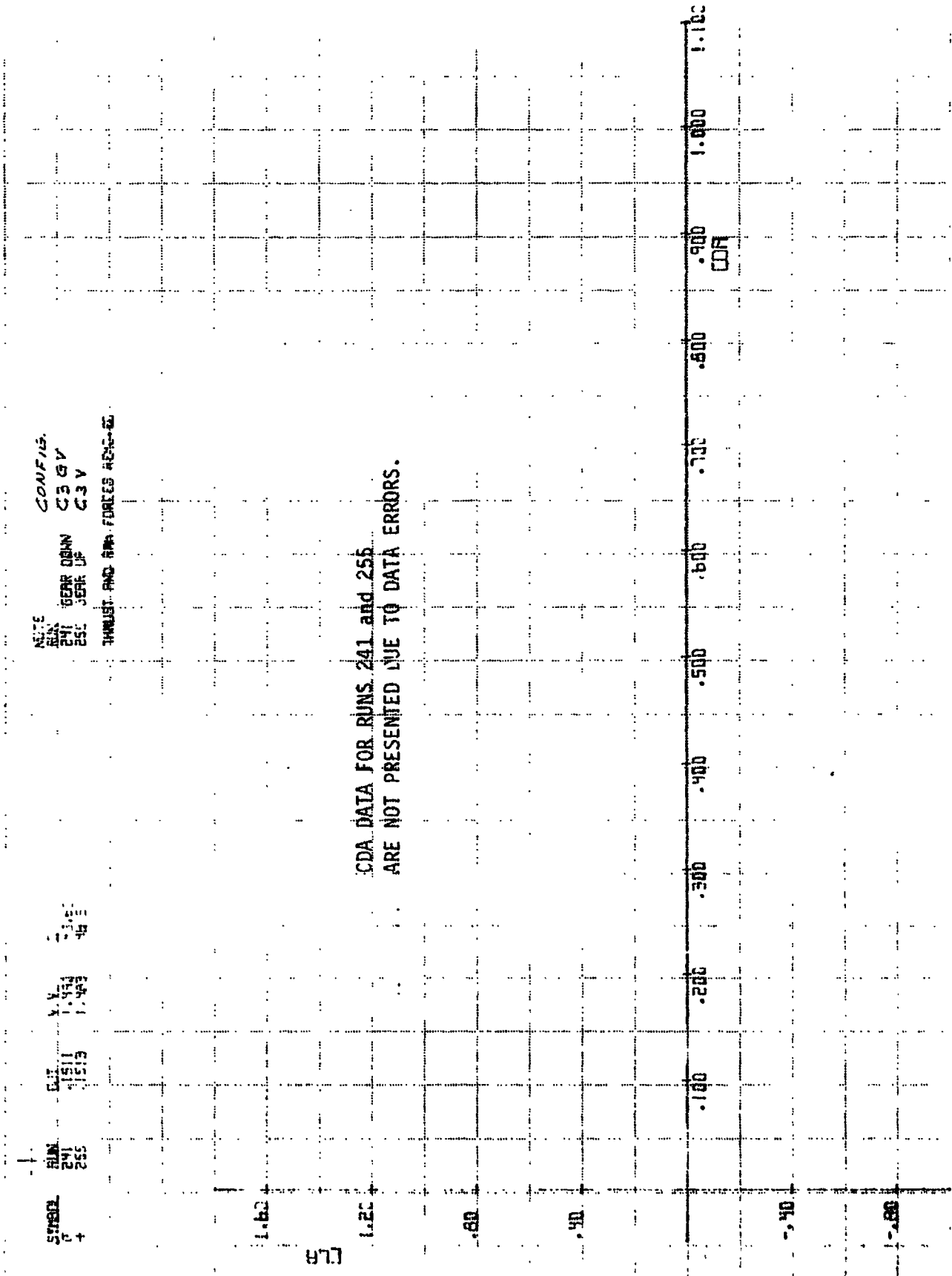
CALL		REVISED	DATE	LPMR VS CLA LTOL CONFIGURATION CHARACTERISTICS LAMBDA = 0. H. TAIL OFF	LENTD 8544 AX-1955 FIG. A. 30 CONT'D PAGE 223
CHECK			12/10/76		
APR					
APP					

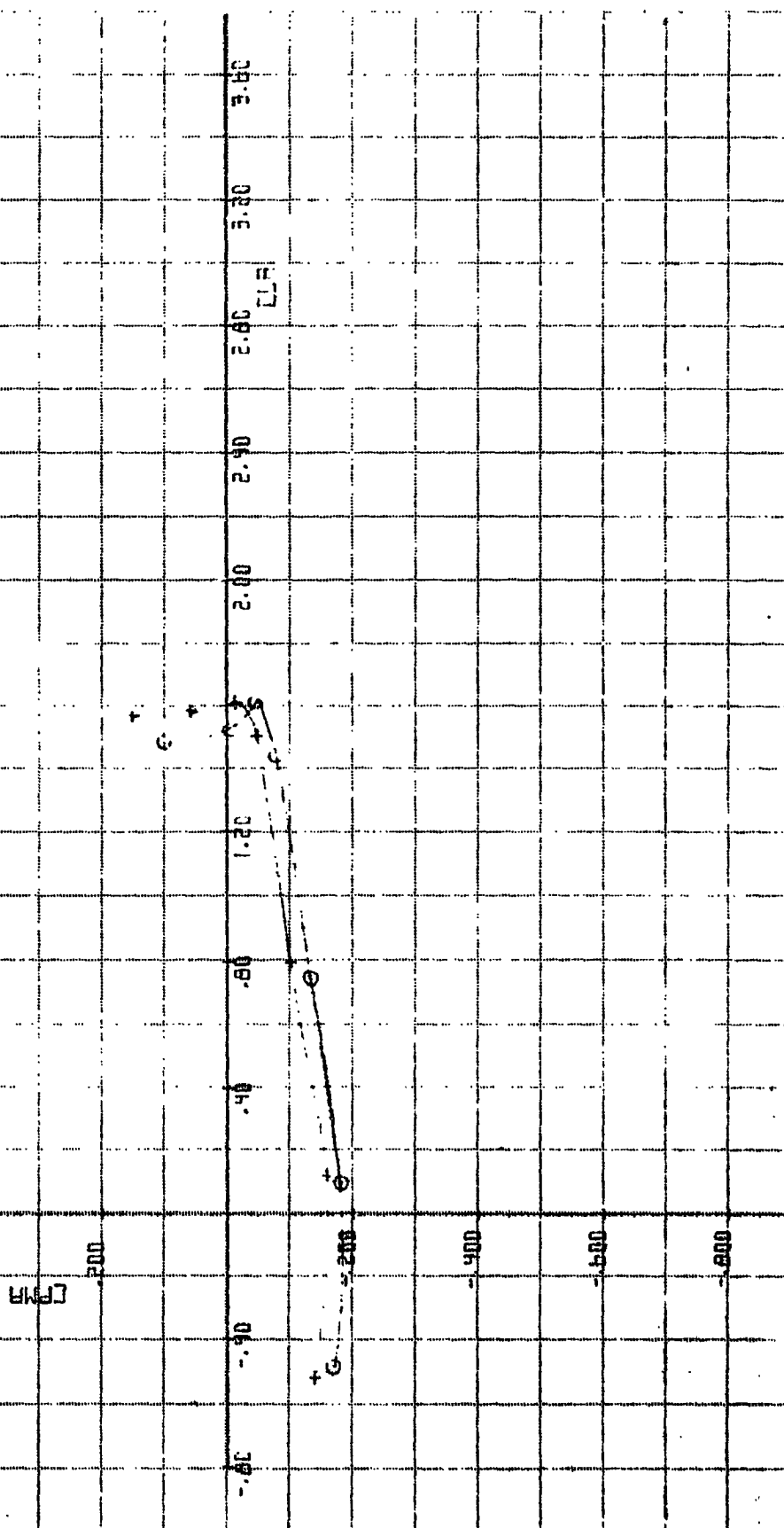
LEAD 2544
RX-1955
FIG. A. 3
CONT'D
PAGE
223

REVISED	DATE	FIG. A.31 CONT'D
1970-12-13		225
THE BOEING COMPANY		

NOTE: CONFID.
RUN: GEAR DOWN C3 GV
END: GEAR UP C3 V
THROUST AND AIR FORCES REMOVED

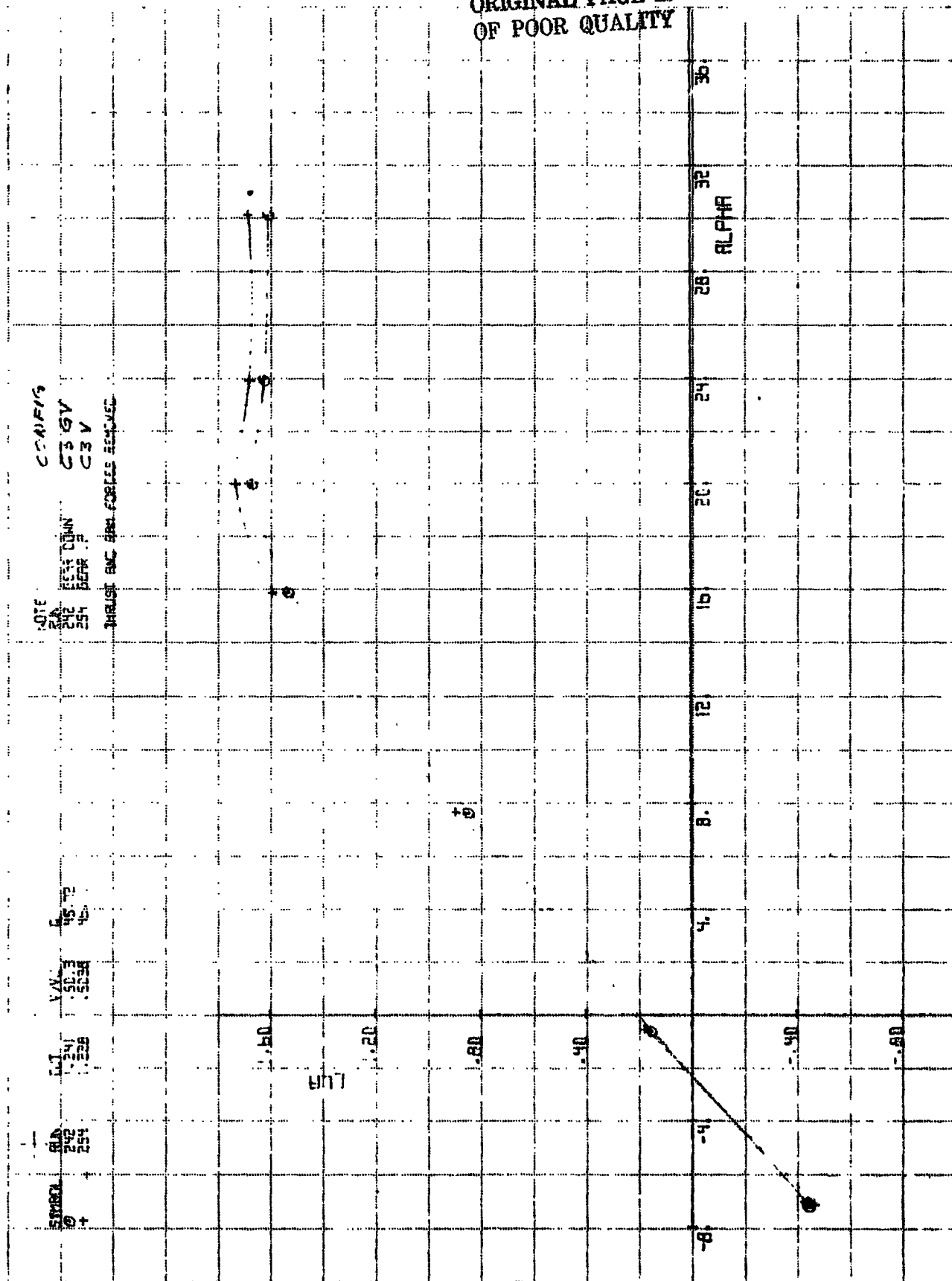
CDA DATA FOR RUNS 241 and 255
ARE NOT PRESENTED DUE TO DATA ERRORS.



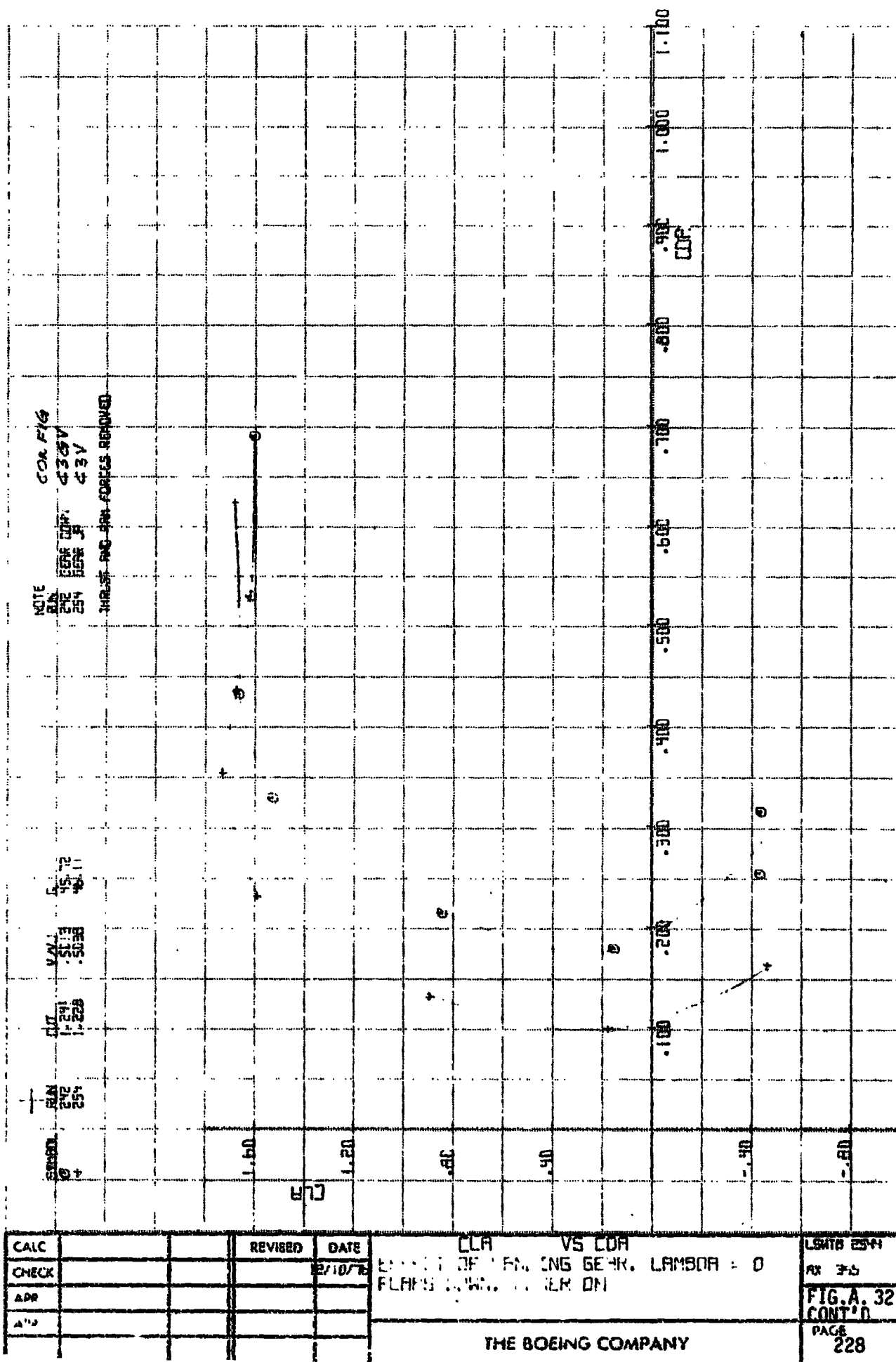
[illegible]

CALC			REVISED	DATE	CPMA VS CLR	LEAD 25-11
CHECK				2/10/76	EFFECT OF LANDING GEAR, LAMBDA = 0	FX-1355
APR					FLIGHT DOWN, POWER OFF	FIG. A.31
APR						CONT'D
					THE BOEING COMPANY	PAGE 226

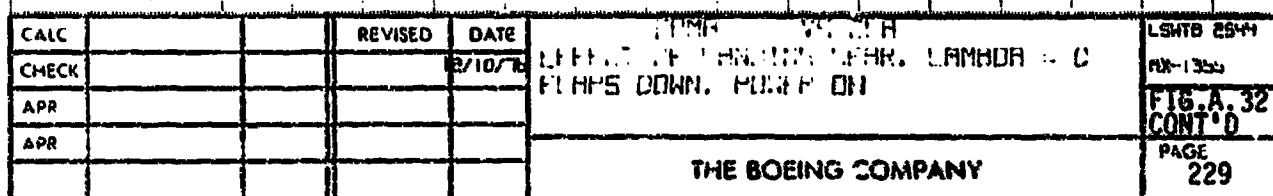
4522 252 310
RECEIVED
FEB 22 1966
C3 V
C3 GV
C3A F15



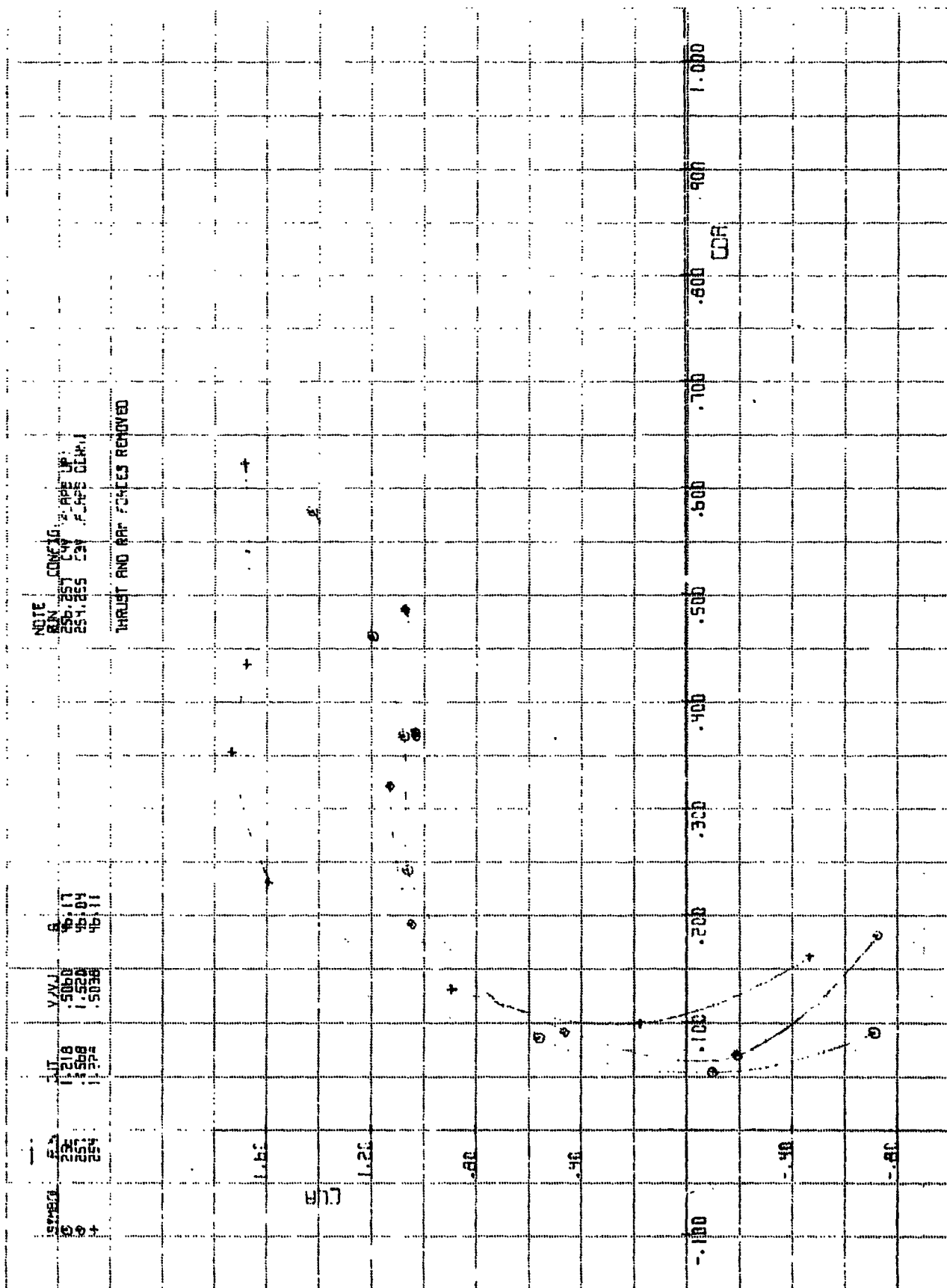
CALC			REVISED	DATE	CLAR VS ALPHA	LSWTD 2544
CHECK				7/10/81	1.00000 OF LAMINATING THICK. LAMBDA = 0	NX 1995
APR					FIBER'S DOWN. TENSILE 100	FIG.A.32
AIR						PAGE 227
					THE BOEING COMPANY	



7-10-68
E105
#22-1
+ 252



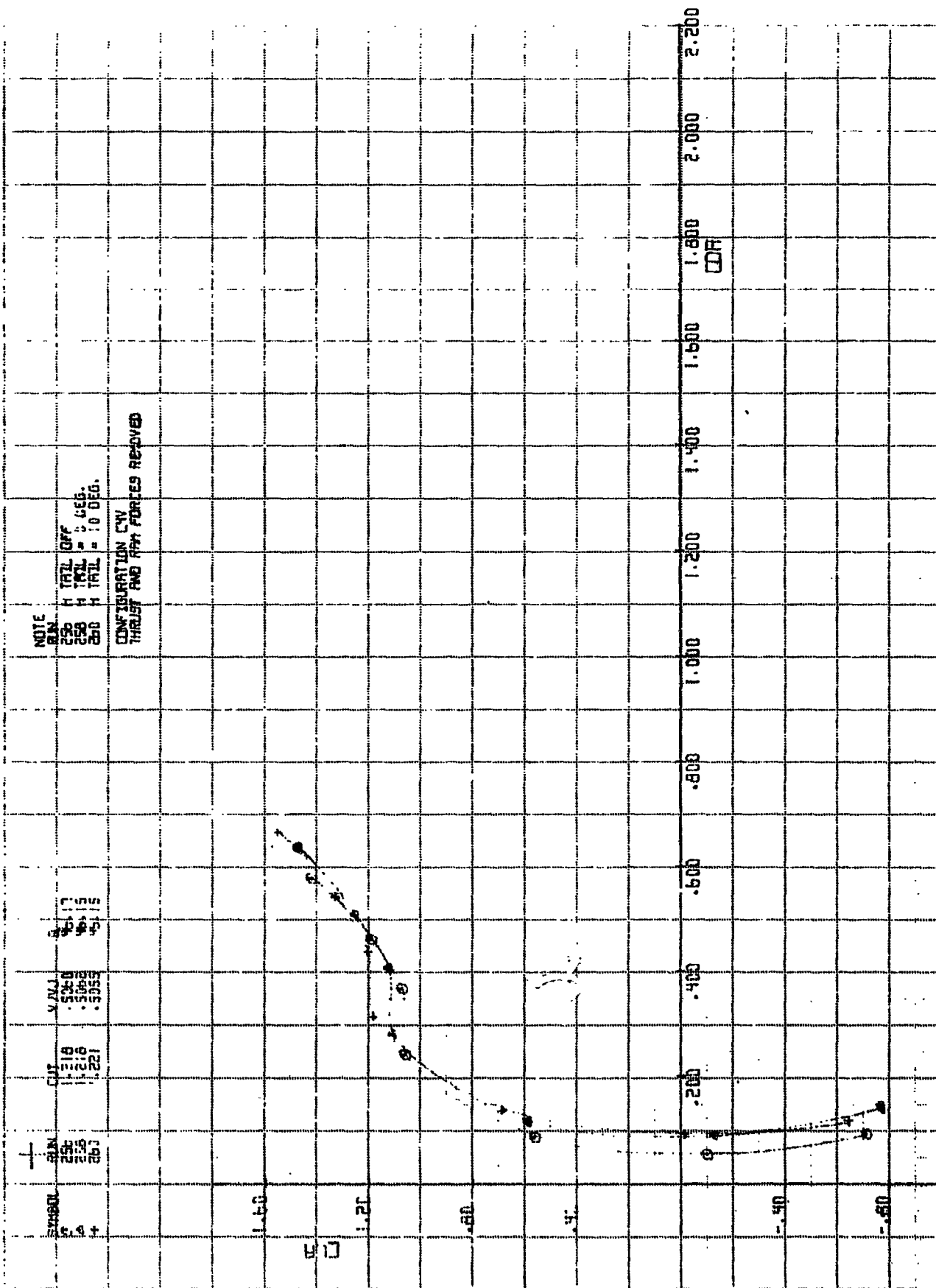
NOTE
 CONFIG. 2: APE UP
 256.25T CAV F. APE CLIM
 251.855 CAV F. APE CLIM
 THRUST AND AIR FORCES REMOVED



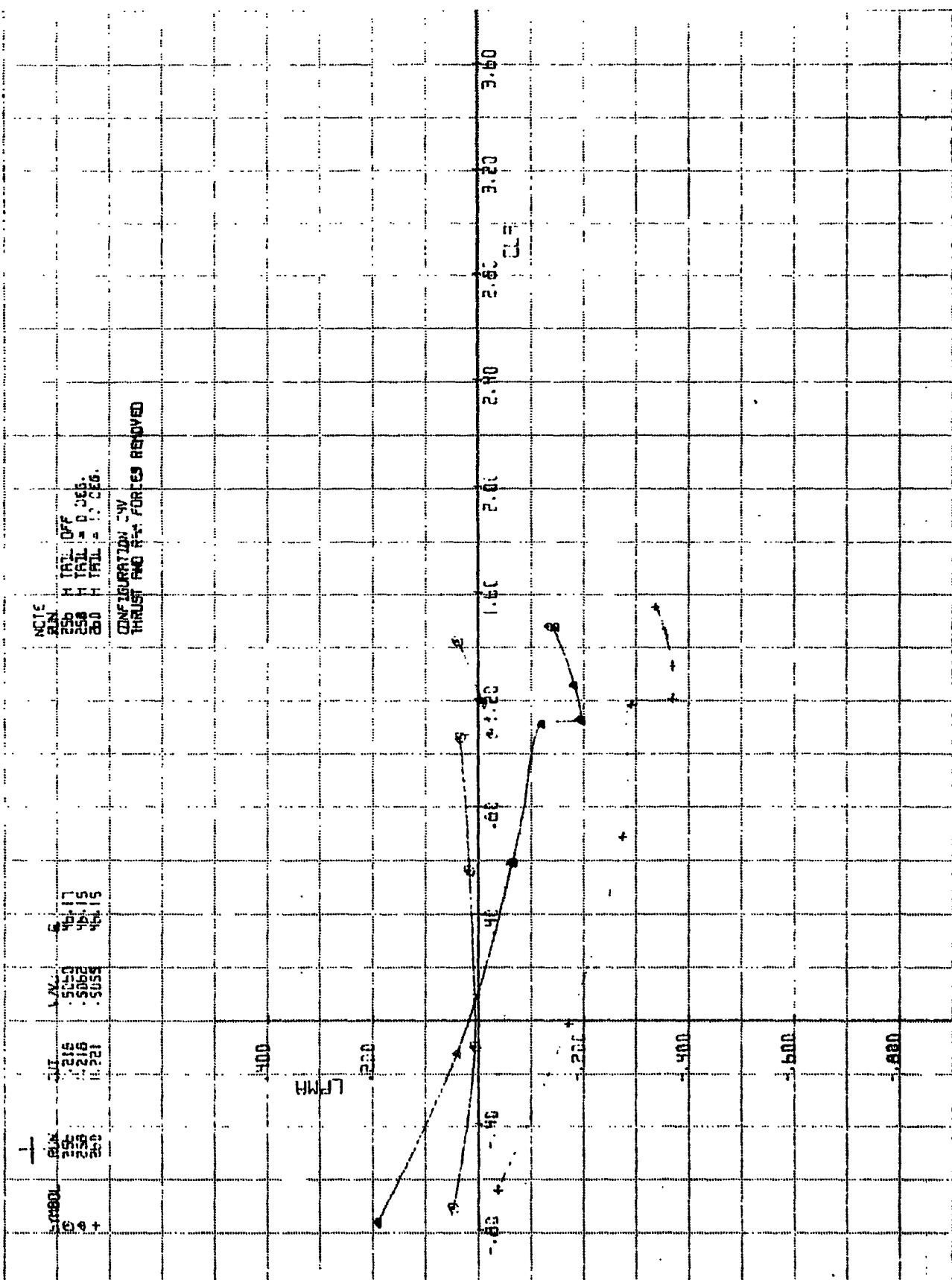
CALC			REVISED	DATE	LCA VS CDA	LSWTB 2544
CHECK				12/10/78	EFFECT OF FLAP AND SLAT	RX-1355
APR					LAMBDA = 0	FIG.A. 33
APR						CONT'D
					THE BOEING COMPANY	PAGE 231

[illegible]

CALC			REVISED	DATE	CLA VS ALPHA	LM70 23-4
CHECK				2/10/8	EFFECT OF HORIZONTAL TAIL	AX-1355
APR					LAMDA = 0 DEG. FLAP'S UP. CJT = 1.2	FIG.A. 34
APR					THE BOEING COMPANY	PAGE 233

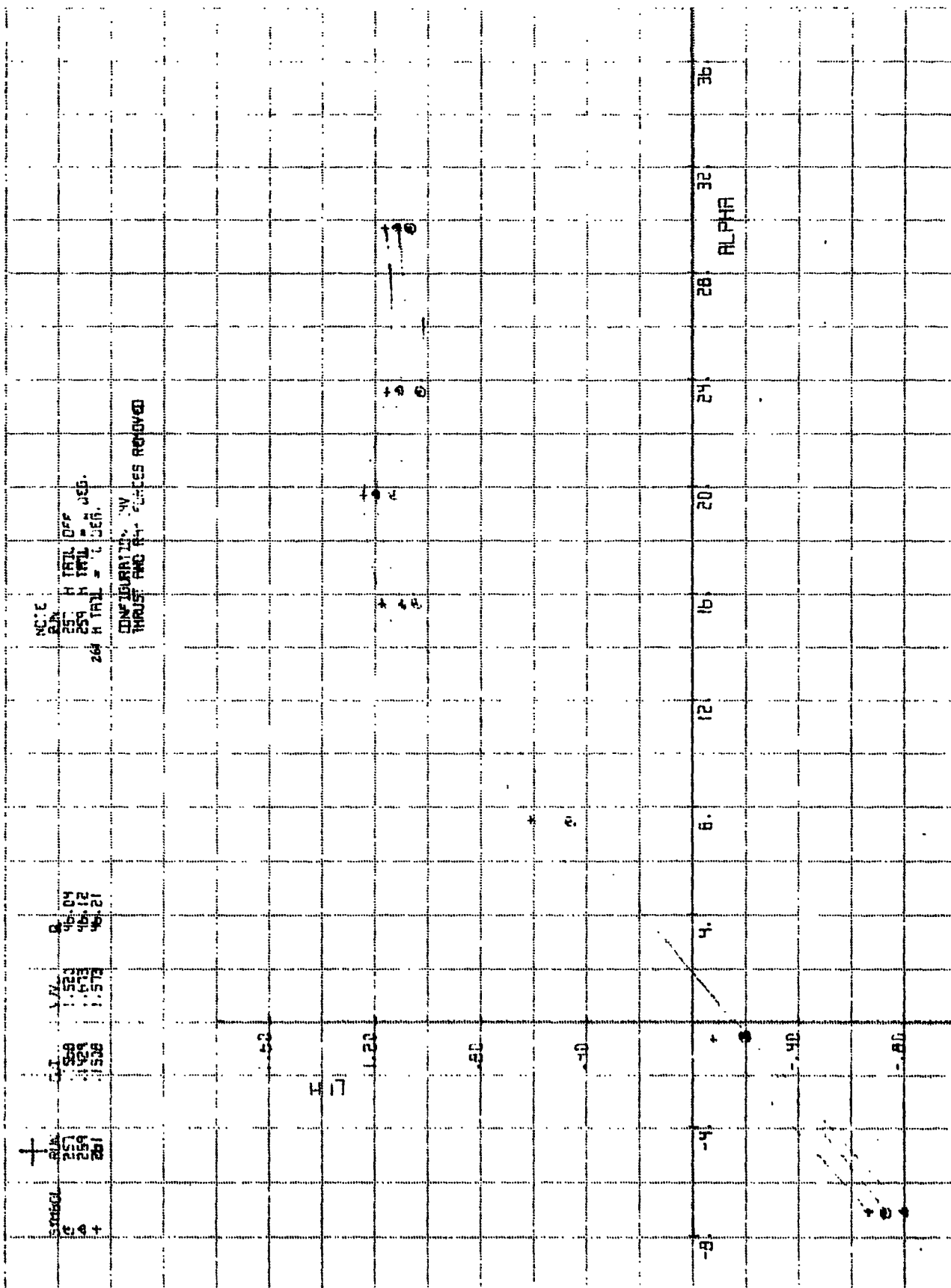


CL	VS CDR	LSMTB 2574
EFFECT OF HORIZONTAL TAIL		AX-1335
THROW = 0 DEG. FLAPS UP, ENT = 1.2		FIG.A. 34
		CONT'D
		PAGE
		234
THE BOEING COMPANY		



CALC		REVISED	DATE	CLMA VS CL EFFECT OF HORIZONTAL TAIL LAMDA = 0 DEG. FLAPS UP. CLT = 1.2	LSWB 25-4 RX-1355 FIG. A. 34 CONT'D PAGE 235
CHECK			2/10/75		
APR					
APR					
THE BOEING COMPANY					

FIG. A. 34
 34.

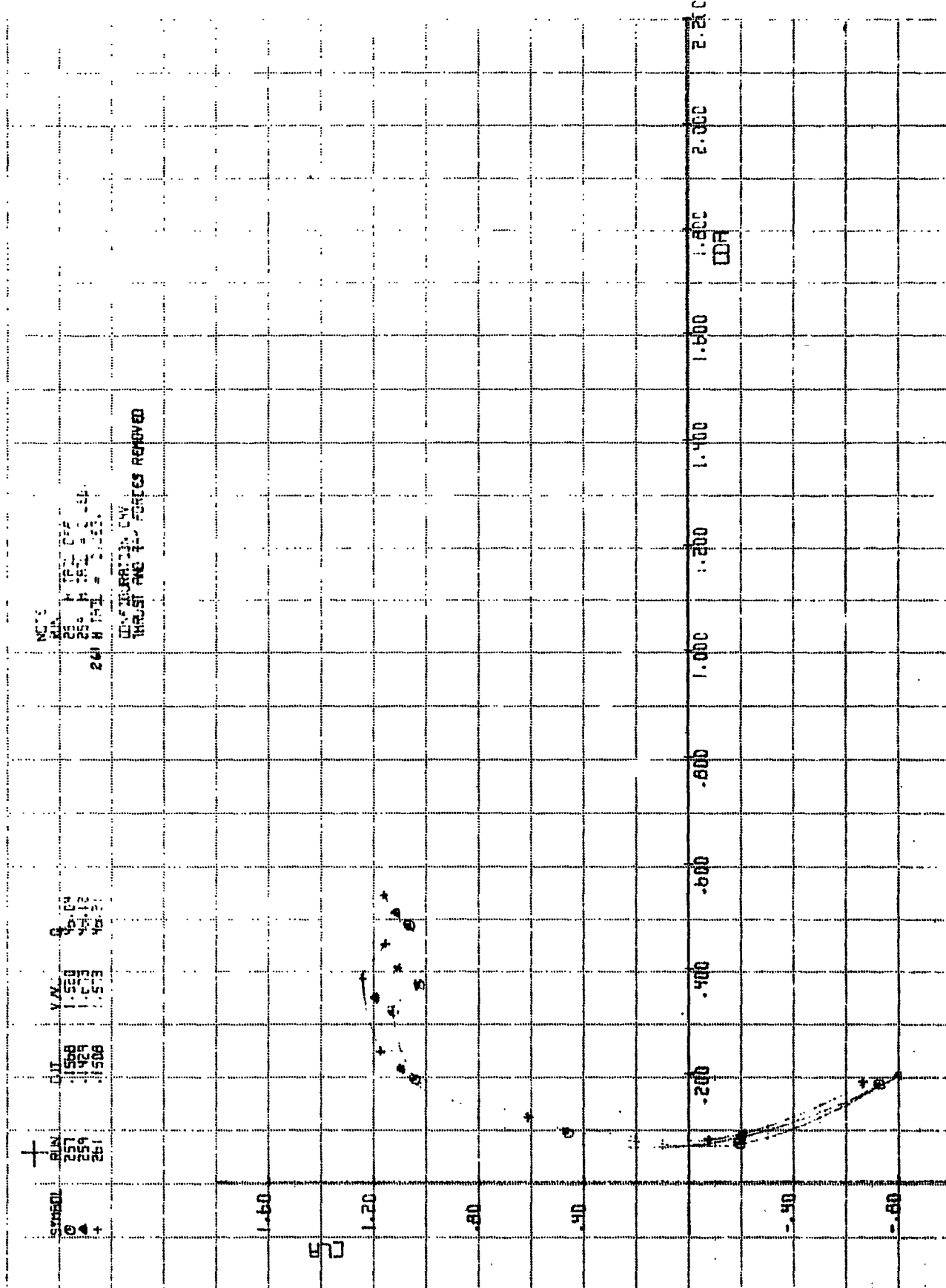


NOTE:
 25 H TAIL DEF
 25 H TAIL DEF
 25 H TAIL DEF
 25 H TAIL DEF

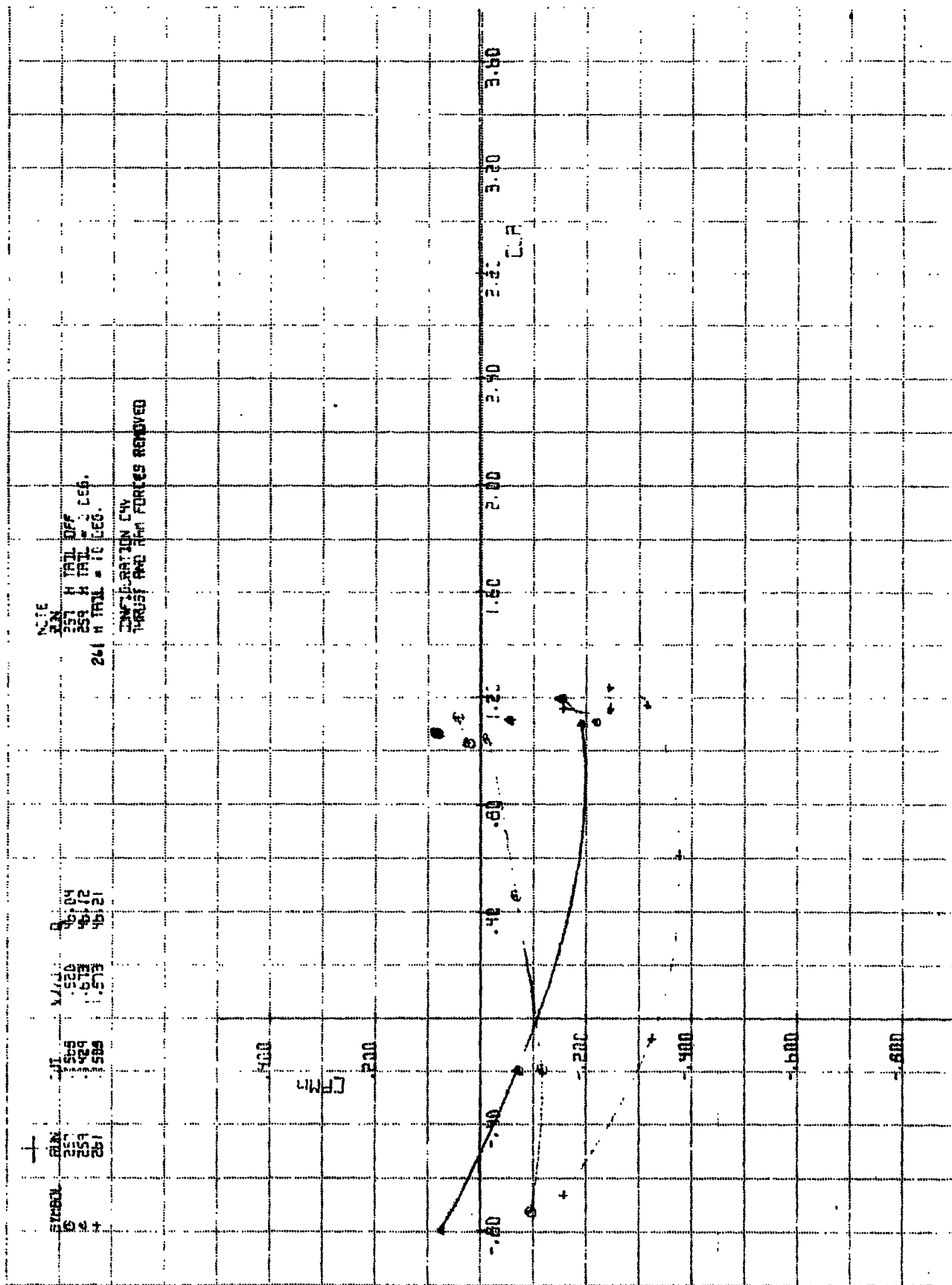
CONFIGURATION, W
 THRUST AND RATES REMOVED

CALC		REVISED	DATE	CLH VS ALPHA EFFECT OF HORIZONTAL TAIL LENGTH - 0.001, ALPHA UP, C.T. = .14	LSMTB 2544
CHECK					HX-1355
APR					FIG. A. 35
APR					PAGE 236

THE BOEING COMPANY



CLC		REVISED	DATE	CLC VS CDR	LSMTD 23-11
CHECK			2/10/76	EFFECT OF HORIZONTAL TH	AX-1355
APR				LENGTH = 0.00, FLAPS UP, LIFT = 1.0	FIG. A. 35
APR					CONT'D
				THE BOEING COMPANY	PAGE 237



CALC		REVISED	DATE	CPMA VS CLA	LEWIS 2544
CHECK			2/10/78	EFFECT OF HORIZONTAL TAIL	PA-1355
APR				LAMDA = 0 DE'S. FLAPS UP. CLT = .18	FIG.A.35
APR					CONT'D
				THE BOEING COMPANY	PAGE 238

NOTE:
 1. F-4P OVER UP: F-3729
 2. F-4P OVER DOWN: F-3710
 CONFIRMATION CASELY
 THROUGH PAC R-4 FORCES REMOVED

1. 1.00
 2. 1.00
 3. 1.00
 4. 1.00
 5. 1.00
 6. 1.00
 7. 1.00
 8. 1.00
 9. 1.00
 10. 1.00
 11. 1.00
 12. 1.00
 13. 1.00
 14. 1.00
 15. 1.00
 16. 1.00
 17. 1.00
 18. 1.00
 19. 1.00
 20. 1.00
 21. 1.00
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 24. 1.00
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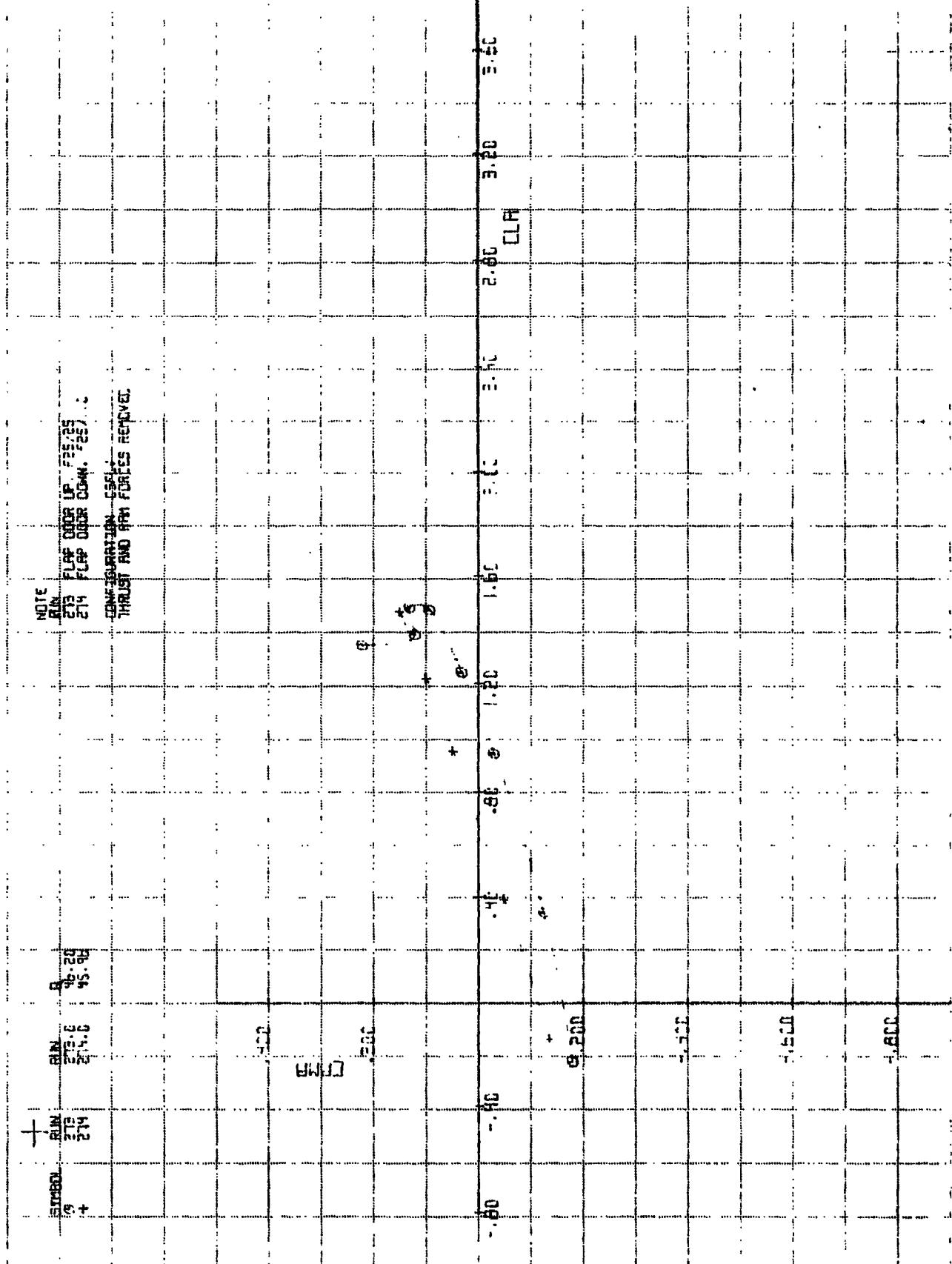
CALC		REVISED	DATE
CHECK			1.1.77
APR			
APR			

CLA VS CDA
 EFFECT OF FLAP DOOR. NAC-L'S OFF

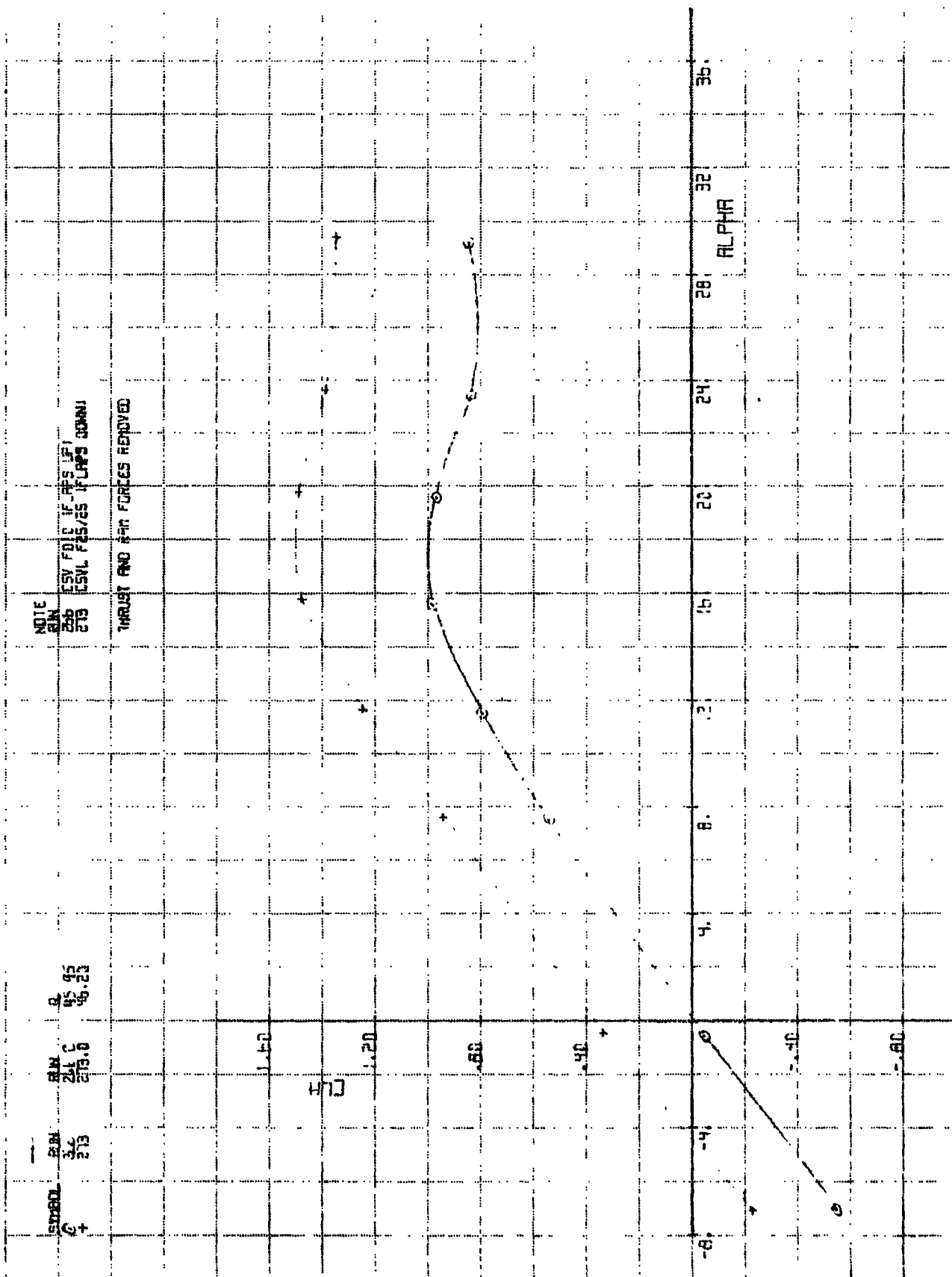
THE BOEING COMPANY

LSMTB 22 1
 AX-1355
 FIG.A.36
 CONT'D
 PAGE 240

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OF POOR QUALITY



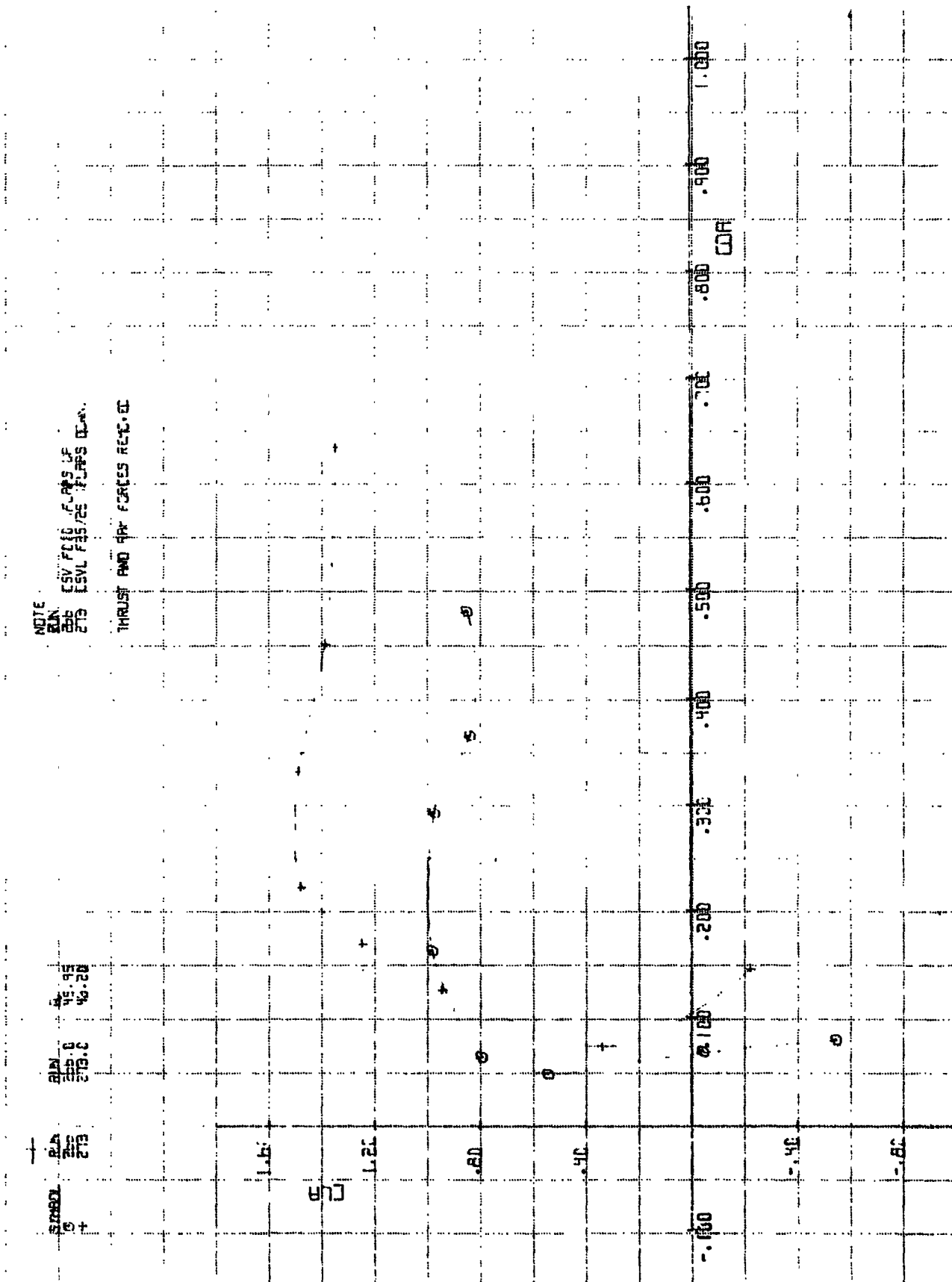
CALC		REVISED	DATE	CLM VS CLP	LSMTB 2544
CHECK			2/10/76	EFFECT OF FLAP ORDER. NAO. LES OFF	AX-132
APP					FIG. A.36
APP					CONT'D
				THE BOEING COMPANY	PAGE 241



CALC		REVISED	DATE	11A VS ALPHA	LSMTB 2541
CHECK			271078	FRONT FLAP AND SLAT	FX 1305
APR				NALL LES LIFT	FIG. A.37
APR					PAGE 242
THE BOEING COMPANY					

NOTE
 RUN CSV FLG FLAPS UP
 273 CSVL FAS/28 FLAPS DOWN
 THRUST AND PW FUEL RES. AT

CALC			REVISED	DATE	CLH VS CDR EFFECT OF FLAP HINDRANCE INCLINED OFF	LSMTB 25-1
CHECK				12/10/78		AX-13.5
APP						FIG. A.37
APP						CONT'D
					THE BOEING COMPANY	PAGE 243



NOTE: CONFIGURATION
 2020347010V INACCESSIBLE OFF
 255 257 259 261 263 265 267 269 271 273 275 277 279 281 283 285 287 289 291 293 295 297 299 301 303 305 307 309 311 313 315 317 319 321 323 325 327 329 331 333 335 337 339 341 343 345 347 349 351 353 355 357 359 361 363 365 367 369 371 373 375 377 379 381 383 385 387 389 391 393 395 397 399 401 403 405 407 409 411 413 415 417 419 421 423 425 427 429 431 433 435 437 439 441 443 445 447 449 451 453 455 457 459 461 463 465 467 469 471 473 475 477 479 481 483 485 487 489 491 493 495 497 499 501 503 505 507 509 511 513 515 517 519 521 523 525 527 529 531 533 535 537 539 541 543 545 547 549 551 553 555 557 559 561 563 565 567 569 571 573 575 577 579 581 583 585 587 589 591 593 595 597 599 601 603 605 607 609 611 613 615 617 619 621 623 625 627 629 631 633 635 637 639 641 643 645 647 649 651 653 655 657 659 661 663 665 667 669 671 673 675 677 679 681 683 685 687 689 691 693 695 697 699 701 703 705 707 709 711 713 715 717 719 721 723 725 727 729 731 733 735 737 739 741 743 745 747 749 751 753 755 757 759 761 763 765 767 769 771 773 775 777 779 781 783 785 787 789 791 793 795 797 799 801 803 805 807 809 811 813 815 817 819 821 823 825 827 829 831 833 835 837 839 841 843 845 847 849 851 853 855 857 859 861 863 865 867 869 871 873 875 877 879 881 883 885 887 889 891 893 895 897 899 901 903 905 907 909 911 913 915 917 919 921 923 925 927 929 931 933 935 937 939 941 943 945 947 949 951 953 955 957 959 961 963 965 967 969 971 973 975 977 979 981 983 985 987 989 991 993 995 997 999 1001 1003 1005 1007 1009 1011 1013 1015 1017 1019 1021 1023 1025 1027 1029 1031 1033 1035 1037 1039 1041 1043 1045 1047 1049 1051 1053 1055 1057 1059 1061 1063 1065 1067 1069 1071 1073 1075 1077 1079 1081 1083 1085 1087 1089 1091 1093 1095 1097 1099 1101 1103 1105 1107 1109 1111 1113 1115 1117 1119 1121 1123 1125 1127 1129 1131 1133 1135 1137 1139 1141 1143 1145 1147 1149 1151 1153 1155 1157 1159 1161 1163 1165 1167 1169 1171 1173 1175 1177 1179 1181 1183 1185 1187 1189 1191 1193 1195 1197 1199 1201 1203 1205 1207 1209 1211 1213 1215 1217 1219 1221 1223 1225 1227 1229 1231 1233 1235 1237 1239 1241 1243 1245 1247 1249 1251 1253 1255 1257 1259 1261 1263 1265 1267 1269 1271 1273 1275 1277 1279 1281 1283 1285 1287 1289 1291 1293 1295 1297 1299 1301 1303 1305 1307 1309 1311 1313 1315 1317 1319 1321 1323 1325 1327 1329 1331 1333 1335 1337 1339 1341 1343 1345 1347 1349 1351 1353 1355 1357 1359 1361 1363 1365 1367 1369 1371 1373 1375 1377 1379 1381 1383 1385 1387 1389 1391 1393 1395 1397 1399 1401 1403 1405 1407 1409 1411 1413 1415 1417 1419 1421 1423 1425 1427 1429 1431 1433 1435 1437 1439 1441 1443 1445 1447 1449 1451 1453 1455 1457 1459 1461 1463 1465 1467 1469 1471 1473 1475 1477 1479 1481 1483 1485 1487 1489 1491 1493 1495 1497 1499 1501 1503 1505 1507 1509 1511 1513 1515 1517 1519 1521 1523 1525 1527 1529 1531 1533 1535 1537 1539 1541 1543 1545 1547 1549 1551 1553 1555 1557 1559 1561 1563 1565 1567 1569 1571 1573 1575 1577 1579 1581 1583 1585 1587 1589 1591 1593 1595 1597 1599 1601 1603 1605 1607 1609 1611 1613 1615 1617 1619 1621 1623 1625 1627 1629 1631 1633 1635 1637 1639 1641 1643 1645 1647 1649 1651 1653 1655 1657 1659 1661 1663 1665 1667 1669 1671 1673 1675 1677 1679 1681 1683 1685 1687 1689 1691 1693 1695 1697 1699 1701 1703 1705 1707 1709 1711 1713 1715 1717 1719 1721 1723 1725 1727 1729 1731 1733 1735 1737 1739 1741 1743 1745 1747 1749 1751 1753 1755 1757 1759 1761 1763 1765 1767 1769 1771 1773 1775 1777 1779 1781 1783 1785 1787 1789 1791 1793 1795 1797 1799 1801 1803 1805 1807 1809 1811 1813 1815 1817 1819 1821 1823 1825 1827 1829 1831 1833 1835 1837 1839 1841 1843 1845 1847 1849 1851 1853 1855 1857 1859 1861 1863 1865 1867 1869 1871 1873 1875 1877 1879 1881 1883 1885 1887 1889 1891 1893 1895 1897 1899 1901 1903 1905 1907 1909 1911 1913 1915 1917 1919 1921 1923 1925 1927 1929 1931 1933 1935 1937 1939 1941 1943 1945 1947 1949 1951 1953 1955 1957 1959 1961 1963 1965 1967 1969 1971 1973 1975 1977 1979 1981 1983 1985 1987 1989 1991 1993 1995 1997 1999 2001 2003 2005 2007 2009 2011 2013 2015 2017 2019 2021 2023 2025 2027 2029 2031 2033 2035 2037 2039 2041 2043 2045 2047 2049 2051 2053 2055 2057 2059 2061 2063 2065 2067 2069 2071 2073 2075 2077 2079 2081 2083 2085 2087 2089 2091 2093 2095 2097 2099 2101 2103 2105 2107 2109 2111 2113 2115 2117 2119 2121 2123 2125 2127 2129 2131 2133 2135 2137 2139 2141 2143 2145 2147 2149 2151 2153 2155 2157 2159 2161 2163 2165 2167 2169 2171 2173 2175 2177 2179 2181 2183 2185 2187 2189 2191 2193 2195 2197 2199 2201 2203 2205 2207 2209 2211 2213 2215 2217 2219 2221 2223 2225 2227 2229 2231 2233 2235 2237 2239 2241 2243 2245 2247 2249 2251 2253 2255 2257 2259 2261 2263 2265 2267 2269 2271 2273 2275 2277 2279 2281 2283 2285 2287 2289 2291 2293 2295 2297 2299 2301 2303 2305 2307 2309 2311 2313 2315 2317 2319 2321 2323 2325 2327 2329 2331 2333 2335 2337 2339 2341 2343 2345 2347 2349 2351 2353 2355 2357 2359 2361 2363 2365 2367 2369 2371 2373 2375 2377 2379 2381 2383 2385 2387 2389 2391 2393 2395 2397 2399 2401 2403 2405 2407 2409 2411 2413 2415 2417 2419 2421 2423 2425 2427 2429 2431 2433 2435 2437 2439 2441 2443 2445 2447 2449 2451 2453 2455 2457 2459 2461 2463 2465 2467 2469 2471 2473 2475 2477 2479 2481 2483 2485 2487 2489 2491 2493 2495 2497 2499 2501 2503 2505 2507 2509 2511 2513 2515 2517 2519 2521 2523 2525 2527 2529 2531 2533 2535 2537 2539 2541 2543 2545 2547 2549 2551 2553 2555 2557 2559 2561 2563 2565 2567 2569 2571 2573 2575 2577 2579 2581 2583 2585 2587 2589 2591 2593 2595 2597 2599 2601 2603 2605 2607 2609 2611 2613 2615 2617 2619 2621 2623 2625 2627 2629 2631 2633 2635 2637 2639 2641 2643 2645 2647 2649 2651 2653 2655 2657 2659 2661 2663 2665 2667 2669 2671 2673 2675 2677 2679 2681 2683 2685 2687 2689 2691 2693 2695 2697 2699 2701 2703 2705 2707 2709 2711 2713 2715 2717 2719 2721 2723 2725 2727 2729 2731 2733 2735 2737 2739 2741 2743 2745 2747 2749 2751 2753 2755 2757 2759 2761 2763 2765 2767 2769 2771 2773 2775 2777 2779 2781 2783 2785 2787 2789 2791 2793 2795 2797 2799 2801 2803 2805 2807 2809 2811 2813 2815 2817 2819 2821 2823 2825 2827 2829 2831 2833 2835 2837 2839 2841 2843 2845 2847 2849 2851 2853 2855 2857 2859 2861 2863 2865 2867 2869 2871 2873 2875 2877 2879 2881 2883 2885 2887 2889 2891 2893 2895 2897 2899 2901 2903 2905 2907 2909 2911 2913 2915 2917 2919 2921 2923 2925 2927 2929 2931 2933 2935 2937 2939 2941 2943 2945 2947 2949 2951 2953 2955 2957 2959 2961 2963 2965 2967 2969 2971 2973 2975 2977 2979 2981 2983 2985 2987 2989 2991 2993 2995 2997 2999 3001 3003 3005 3007 3009 3011 3013 3015 3017 3019 3021 3023 3025 3027 3029 3031 3033 3035 3037 3039 3041 3043 3045 3047 3049 3051 3053 3055 3057 3059 3061 3063 3065 3067 3069 3071 3073 3075 3077 3079 3081 3083 3085 3087 3089 3091 3093 3095 3097 3099 3101 3103 3105 3107 3109 3111 3113 3115 3117 3119 3121 3123 3125 3127 3129 3131 3133 3135 3137 3139 3141 3143 3145 3147 3149 3151 3153 3155 3157 3159 3161 3163 3165 3167 3169 3171 3173 3175 3177 3179 3181 3183 3185 3187 3189 3191 3193 3195 3197 3199 3201 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THE HIGHEST AND FURTHEST REACHED

X	23	154	145	4546
X	23	154	145	4546



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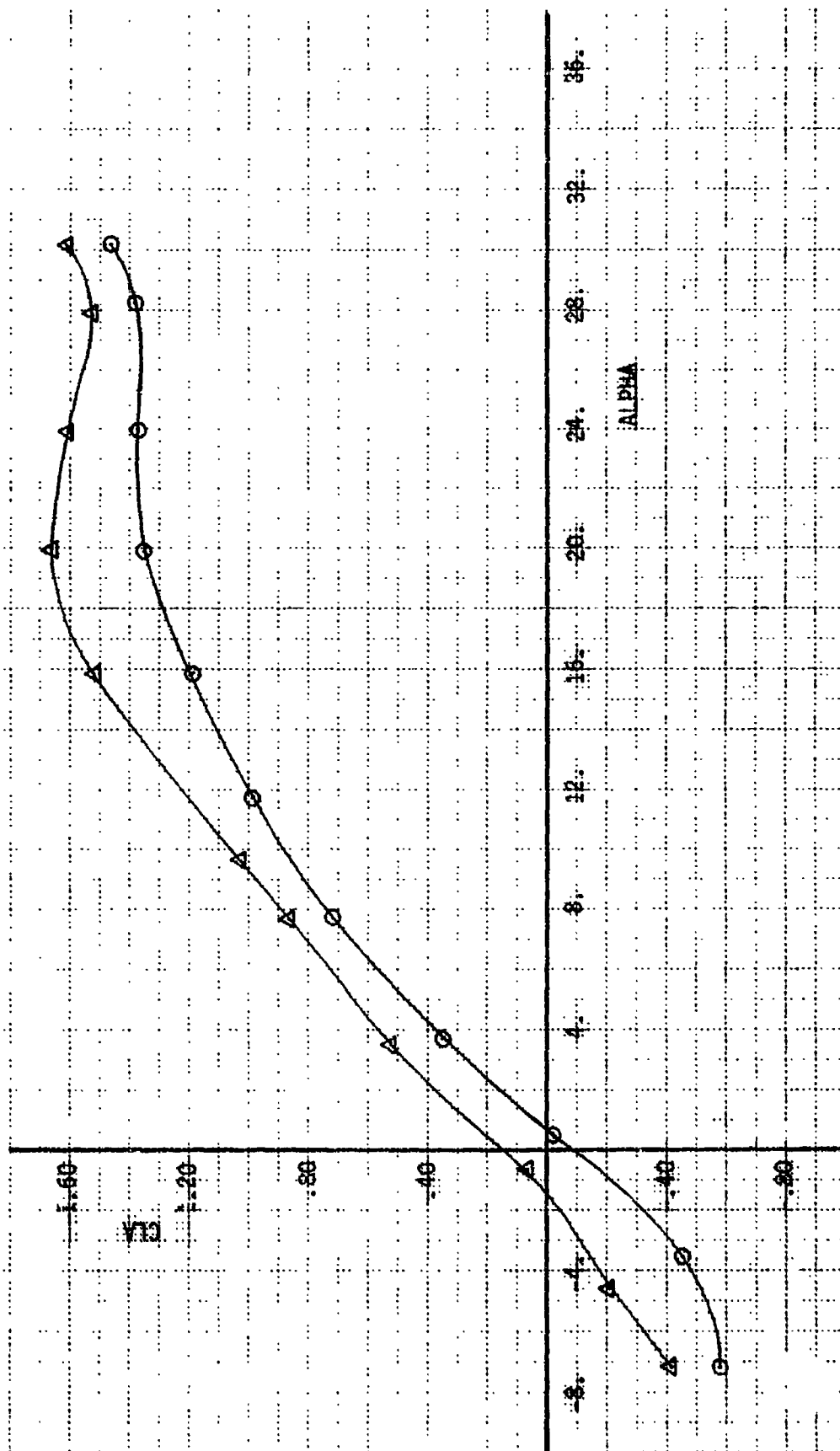
FIG. A.38
CONT'D

[illegible]

CALC			REVISED	DATE	CPMA VS CLA	LSMTB 2544
CHECK				2/10/76	CONFIGURATION BUI' UP- POWER OFF	AX-1355
APR						FIG.A.38
APR						CONT'D
					THE BOEING COMPANY	PAGE 247

NOTE
CONFIGURATION C2GV
THRUST AND RAM FORCES REMOVED

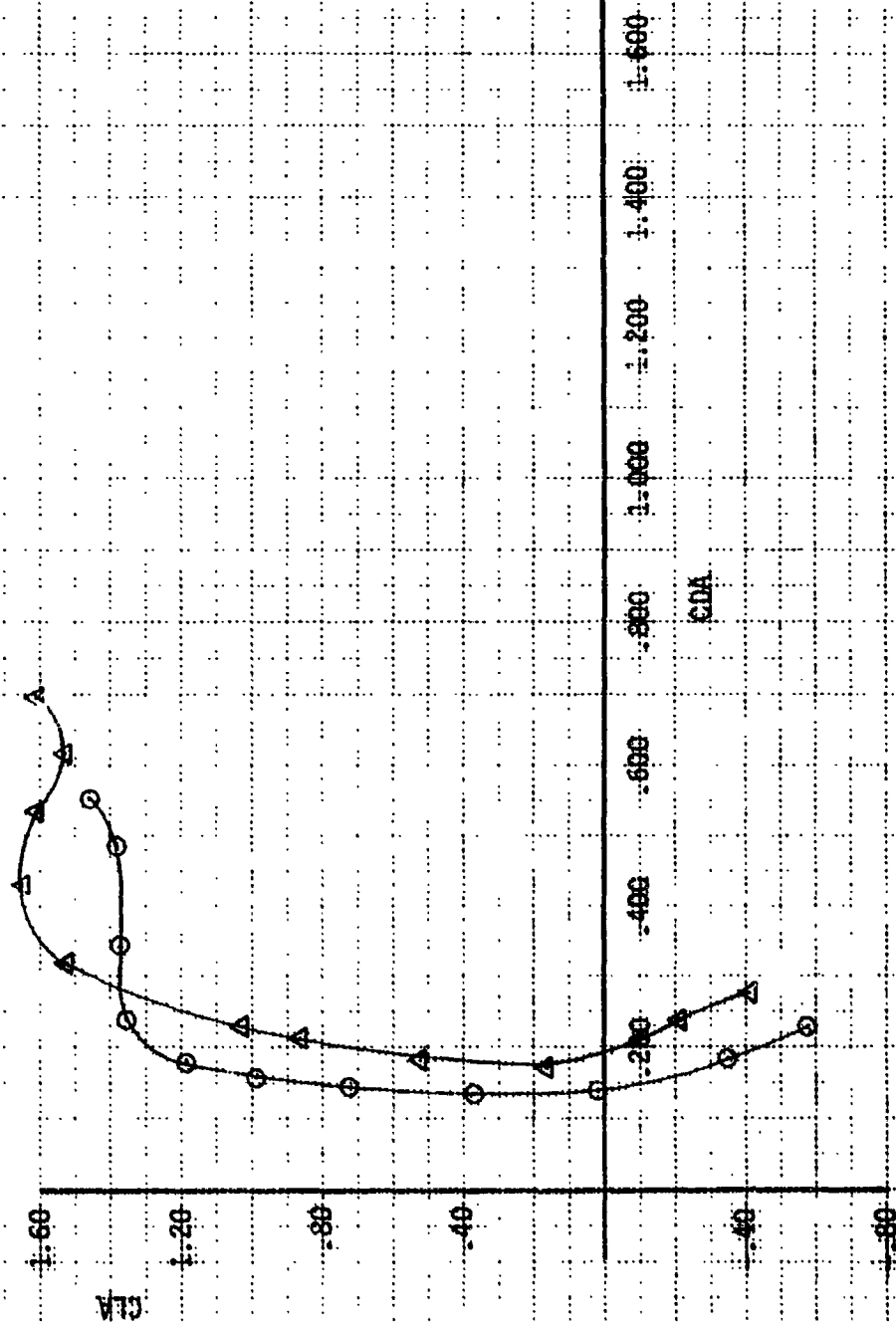
SYMBOL	RUN	CJA	CJB	CJC	CJT	Q
O	229	.62	.61	.22	1.45	46.7
Δ	230	.64	.64	.00	1.28	45.6



CALL		REVISED	DATE	EFFECT OF NOSE FAN THRUST VARIATION LAMBDA = 0°, H. TAIL OFF	FIG. A.39
CHECK					
APR				THE BOEING COMPANY	PAGE 248
APR					

NOTE
CONFIGURATION C26V
THRUST AND RAM FORCES REMOVED

SYMBOL	RUN	CJA	CJB	CJC	CJT	Q
○	229	.62	.61	.22	1.45	46.7
△	230	.64	.64	.00	1.28	45.6



CALC			REVISED	DATE
CHECK				
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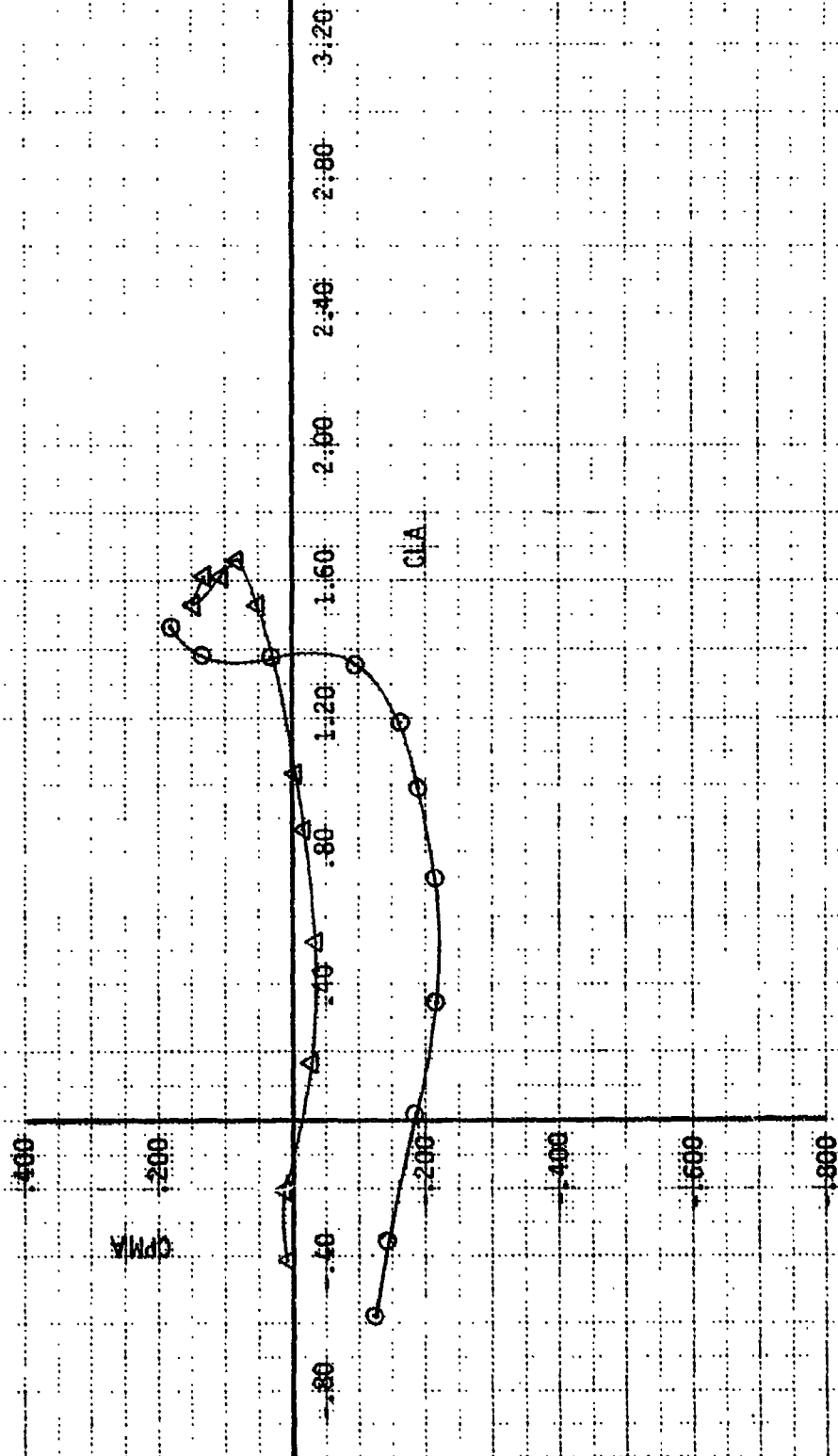
EFFECT OF NOSE FAN THRUST VARIATION
LAMBDA = 0°, H. TAIL OFF

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FIG. A.39
CONT
PAGE 249

NOTE
 CONFIGURATION C2GV
 THRUST AND RAM FORCES REMOVED

SYMBOL	RUH	CJA	CJB	CAC	CJT	Q
O	229	.62	.61	.24	1.45	46.7
A	230	.64	.64	.00	1.28	45.6



CALC	REVISD	DATE
CHECK		
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FIG. A.39
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 PAGE
 250

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Appendix B
THRUST CALIBRATION

Nomenclature for Appendix B

A_{HI}	Inlet Highlight Area
A_{TH}	Inlet Throat Area
C_{VF}	Fan Nozzle Velocity Coefficient
C_{DF}	Fan Nozzle Flow Coefficient
C_{D2}	Inlet Flow Coefficient
$\frac{FG}{\delta_{AMB}}$	Corrected Gross Thrust
M_{PR}	Primary Mass Flow
P_{∞}, P_0	FSC Chamber (Free Stream) Static Pressure
P_{T2}	Inlet Total Pressure
P_{S2}	Inlet Static Pressure
P_{T3}	Fan Nozzle Total Pressure
P_{S3}	Fan Nozzle Static Pressure
P_{T4}, P_{TPR}	Primary (Core) Nozzle Total Pressure
P_{T5}	Tip Turbine Discharge Total Pressure
T_{TPR}	Primary (Core) Nozzle Total Temperature
$\frac{RPM}{\sqrt{\theta_{AMB}}}$	Corrected Speed
V_{FS}	Free Stream Velocity
x_B	Height of Distortion-Generating Blade Above Inlet Highlight

APPENDIX B

THRUST CALIBRATIONS

Thrust calibrations of the turbopowered simulators were conducted in the Boeing Flight Simulation Chamber (FSC), and with the simulators installed in the model during the static ground tests and during the wind tunnel test. The initial calibration conducted in the Flight Simulation Chamber provided the effects of nozzle back pressures above and below ambient pressure, thus simulating ground effects, nozzle back pressuring due to thrust vectoring in excess of 90° to the free stream, and ram conditions in forward flight. The trends obtained from this calibration were coupled with absolute levels obtained from in-place calibrations during the wind tunnel test. The FSC calibrations related the thrust, as measured by strain gauge balances, to: ideal thrust calculated from fan exit static and total pressure and total temperature; fan RPM; and fan tip-drive air supply pressure and temperature. During static tests the relationship of thrust versus RPM was used in determining thrust. Because of expected inlet total pressure distortion in the wind tunnel for certain combinations of inlet angles of attack, forward speed, and power setting, the thrust was determined from fan exhaust pressure and temperature measurements during wind tunnel tests. Thrust versus tip-drive supply pressure served as a back-up method of determining thrust.

Model Description

A schematic of the lift/cruise fan nacelle is shown in Figure B.1. The turbopowered simulators were 14 cm (5.5 inches) Technology Development Inc. tip-turbine driven fans supplied by NASA for this test. The model scale was thus selected to fit the available fans. A fan map and a thrust calibration with ambient back pressure to the fan (zero convergence fan nozzle) were available from NASA CR-2051 (Reference 4). It was decided to operate the model fans at the pressure ratio of the "study" engine Allison PD370-16, a T701 derivative (Reference 8). Since thrust data were available only for zero fan back pressure, the nozzle area was set equal to fan flow area plus 2% allowance for boundary layer growth. The model scale was therefore defined by the square root of the ratio of model thrust produced at that exit area to full scale

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thrust, with nozzle pressure ratios reproduced in the model.

External lines of lift-cruise nacelles were scaled from a drawing of the NASA V/STOL technology demonstrator (Reference 9) modified in a few areas as discussed below. The inlet was patterned after a 0.9 scale lift/cruise inlet model tested in support of the V/STOL program (Reference 10). The internal lines of the 0.9 scale model were non-axisymmetric, with the contraction ratio higher on the bottom of the inlet (with the nacelle in the forward thrust position). The 0.094 scale powered model, fabricated for the present test program, required about 20% less inlet area because of the tip turbine flow. The throat was therefore reduced to maintain the throat Mach number and thereby maintain the inlet's ability to operate at high angles of attack and low mass flows. The reduced throat area requirement of the model coupled with the scaled, large highlight allowed the selection of an axisymmetric inlet with a contraction ratio $A_{HI}/A_{TH} = 2.0$ which exceeds that on the bottom of the non-axisymmetric full scale inlet (1.76). The coordinates of the powered lift/cruise model inlet are shown on Figure B.2. The outer wall of the fan nozzle of the lift/cruise nacelle was axisymmetric. The centerbody was designed with an elliptical cross-section assuming that the engine and airplane accessories would be distributed between the top and bottom of the core engine. This design reduces the internal nozzle curvatures and also provided a representative flow area in the vertical plane for thrust vectoring vanes. The fan duct was held at nearly constant area, increasing only 2% to the nozzle exit plane. The fan nozzle of the PD370-16 engine being simulated would be a variable area nozzle and actually divergent at low speeds. Thrust vectoring vanes were installed in the lift-cruise fan nozzle exits to provide yaw control when the nacelles are tilted 90° for lift production.

The primary nozzles of the lift/cruise engines had scaled exit areas and were slightly convergent, although the PD370-16 primary nozzle would be divergent. The model nozzle was supplied with high pressure air off the turbine air supply which was reduced to the required pressure through choke plates and screens.

The inlet of the nose fan was designed as a short non-axisymmetric bellmouth. The contraction ratio of the bellmouth varied from 1.76 on the forward part to 1.4 on the sides and aft portion. Figure B.3 shows several sections of the inlet. The nozzle of the nose fan was a cylindrical duct cut off at 15° to its axis. The fan hub was not faired out but left as a base in the nozzle exit plane. Like the lift/cruise fans, the nose fan was fitted with vanes for yaw control. A schematic of the model forebody is shown in Figure B.4. It should be noted that the upper half of the forebody was left on the fan simulator during calibrations in the flight simulation chamber.

The model nozzles were instrumented with total pressure rakes, static pressure taps and thermocouples to provide the required data to define their velocity and flow coefficients. The lift/cruise inlets were instrumented with static taps for flow calibration and one total pressure rake to detect inlet flow separation.

The schematics of the instrumentation are shown on Figure B.1 and depicted in Figure B.5. The outer probes in all fan nozzle rakes were placed in the tip turbine stream and manifolded together. The remaining probes were read individually on nacelle "B" and the nose fan "C", but manifolded by rake in nacelle "A". All statics were read individually.

Flight Simulation Chamber

A schematic of the Flight Simulation Chamber (FSC) is shown on Figure B.6. Air is drawn through the model into a stilling chamber, then through one or a set of critical flow venturis and finally exhausted to the atmosphere by an ejector. The model is mounted on a balance where thrust is measured. Airflow is measured by the Multiple Critical Flow Venturis (MCV). The chamber pressure may be maintained below ambient by proper selection of venturis and suction of the ejector, thus simulating the ram effect encountered in a wind tunnel or in flight. With powered models, the chamber may also be maintained at a pressure above ambient, simulating back pressurization due to ground proximity, jet interference, tailwind, etc.

A total pressure probe on an X-Y translating mechanism was mounted inside the FSC and used to survey the primary nozzle flow.

Test Procedure

Airflow through the FSC is a function of the selected venturi size and the total pressure within the chamber. The model was to be calibrated over a range of back pressures equivalent to speeds up to 77 meters/sec (150 knots), and over a range of power settings from near zero thrust to about 85% RPM. Data from Reference 4 indicated that the fan nozzle airflow at that RPM would be about 2.72 kg/sec (6 lb/sec). The test was therefore conducted by varying airflow from 2.72 kg/sec (6 lb/sec) to about 0.91 kg/sec (2 lb/sec) in 0.23 kg/sec (1/2 lb/sec) increments by installing appropriate multiple critical Venturi Combinations (MCV). At each MCV setting, fan RPM was varied over the required range to produce an FSC internal pressure from 5% below to 5% above ambient, corresponding to the ram effect at 82 meters/sec (160 knots). Generally, five back pressure levels were set: $P_{FSC}/P_{ATMO} = .95, .975, 1.0, 1.025, 1.05$. Occasionally, fan stall was encountered at higher back pressures and some test points were omitted. Generally, three scans of data were taken at each test point. A warmup run preceded data runs, because air at 71°C (160°F) was required to drive the tip-turbines and avoid icing of fan shroud, blade tips and stators.

Calibrations

Nacelle pressure instrumentation was routed out of the FSC to a bank of scannivalves and the output was recorded by the BTWT Astrodata System. FSC pressure measurements were routed directly to dedicated transducers. All pressure transducers and the entire electronic system were periodically calibrated by applying a pressure balanced by a deadweight tester to the transducer and comparing the output to the deadweight tester. Airflow measurements were made using calibrated Multiple Critical Flow Venturis (MCV's). The MCV's trace their calibration to the Colorado Engineering Experiment Station.

Thermocouples were compared to a standard thermometer prior to running, when the nacelle and chamber were at a steady temperature. The force measuring system included two balances, bellows to bridge the high pressure air supply to the metric side and pressure compensating bellows to reduce tare forces on the metric portion of the FSC. Balance calibrations were checked by hanging weights and bellows pressure tares were obtained by repeating balance calibrations with pressure applied to the bellows. Momentum tares as well as the proper operation of the pressure and flow measurement system were checked by running a known reference nozzle prior to installation of the model. The indicated flow coefficient of the known nozzle checks the FSC pressure, temperature and flow measuring system, while the indicated thrust coefficient provides the momentum tare, if any.

Velocity and Flow Coefficients

Equations for fan velocity coefficient, fan flow coefficient and inlet flow coefficient are given below, starting from averaged pressure measurements.

$$C_{VF} = \frac{F_{GTOT} - F_{PR}}{\dot{M}_{TT}V_{ITT} + \dot{M}_F V_{IF}} \quad B.1$$

where:

F_{GTOT}	=	total thrust from balance
F_{PR}	=	calculated primary thrust
\dot{M}_{TT}	=	tip turbine mass flow
V_{ITT}	=	tip turbine ideal velocity = $f(P_{TTT}, P_0, T_{TTT})$
\dot{M}_F	=	fan mass flow
V_{IF}	=	fan ideal velocity = $f(P_{TF}, P_0, T_{TF})$
P_{TTT}, P_{T5}	=	average tip turbine exit total pressure (6 manifolded probes)
T_{TTT}	=	tip turbine exit total temperature

T_{TF} = fan exit total temperature (average of 4)

P_{TF}, P_{T3} = average fan exit total pressure For the nose fan and one L/C fan it is the area weighted average of 24 probes; for the other L/C fan it is the area weighted average of 6 rakes, each a manifold of 4 probes).

P_o, P_{∞} = FSC chamber (freestream) static pressure

The primary thrust was obtained from a translating probe survey at the primary nozzle exit plane, from which the mass flow and thrust were calculated, and correlated with measured nozzle pressure ratio. The calculated mass flow of the primary was subtracted from the total measured supply flow to obtain the tip turbine flow. Subsequently, primary thrust was calculated by applying the above mass flow split to the measured supply flow and multiplying it by the ideal velocity based on measured primary total pressure and a velocity coefficient which relates the manifolded primary total pressure rake to the average pressure obtained by the survey.

$$F_{PR} = K_{PR} \times W_{MCV-2} \times V_{IPR} \times C_{VPR} \quad B.2$$

where:

K_{PR} = fraction of total supply flow to primary

W_{MCV-2} = total supply flow

V_{IPR} = ideal velocity = $f(P_{TPR}, P_o, T_{TPR})$

$C_{VPR} = \frac{V_{IPR}}{\text{ideal velocity based on probe survey of exit area}}$

The fan mass flow was measured as the difference of total mass flow drawn out of the FSC through its bank of critical flow Venturis and the total supply flow to the tip turbine and primary.

$$M_F = W_{MCV-4} - W_{MCV-2} \quad (\text{see Figure B.6}) \quad B.3$$

The fan flow coefficient was defined in an unusual manner due to the fact that the fan nozzle also passed the tip turbine flow, and because it was related to internal nozzle static pressure rather than ambient pressure. The tip turbine flow and the geometric area of the tip turbine including the fan shroud were excluded from the definition because the tip turbine flow could

be measured during the wind tunnel test. The average static pressure just inside the nozzle was used to extend the usefulness of the static flow coefficient to wind-on conditions where the nozzle exit plane is subjected to significant local pressure fields created by the airframe and ground proximity.

$$C_{DF} = M_F / M_{IF} \quad B.4$$

where:

$$M_{IF} = f(P_{TF}, P_{SF}, T_{TF}, A_F)$$

$$P_{SF} = \text{average fan nozzle exit static pressure}$$

(L/C fans: 4 taps, nose fan: 6 taps)

$$A_F = \text{flow area from rotor hub to fan shroud}$$

The fan duct and nozzle were designed at a constant area equal to the fan exit area, plus 2% for boundary layer growth. Thus, using the fan rotor flow area was equivalent to using the fan nozzle exit area less the tip turbine and shroud area.

The third coefficient calculated was an inlet flow coefficient.

$$C_{D2} = M_F / M_{I2} \quad B.5$$

where:

$$M_{I2} = \text{ideal inlet flow} = f(P_{T2}, P_{S2}, T_{T2}, A_2)$$

$$P_{T2} = \text{inlet total pressure} = \text{atmospheric pressure}$$

$$P_{S2} = \text{inlet static pressure (average of 4 taps)}$$

$$T_{T2} = \text{inlet total temperature}$$

$$A_2 = \text{geometric flow area in the plane of the static pressure taps}$$

Results

Velocity coefficients and flow coefficients for lift/cruise fans "A" and "B" and lift fan "C" with yaw vanes set at 0° as determined in the flight simulation chamber are shown on Figures B.7-B.12. The data are plotted vs. fan total pressure ratio rather than the more common fan nozzle total to ambient

static pressure ratio. This was done because, in fan powered models, nozzle coefficients are strongly affected by fan power setting and back pressure ratio (P_{T2}/P_{S3}) through their effect on total pressure distribution at the nozzle charging station. The data trends in Figures B.7-B.12 were subsequently used to calculate gross thrust and ram drag from pressure and temperature measurements in the wind tunnel. A correction to the velocity coefficients was applied based on results of in-place calibrations in the tunnel using the model internal balance.

Velocity and flow coefficient curves for back pressures above inlet pressure ($P_{T2}/P_{S3} < 1.0$) often drop or rise steeply at low fan power settings, see Figures B.7 to B.12. This is due to fan stall and the ensuing increase in P_{T3} distortion. When distortion increases, the calculated average total pressure P_{T3} may be biased above or below the true effective total pressure.

Inlet distortion generated by a sharp blade across one inlet is shown on Figure B.13. The blade was positioned such that its upper edge was 2.34 cm (.92") above the inlet highlight. The inlet rake was completely in the wake of the blade, showing a recovery of .85. This inlet distortion produced a fan exit total pressure distortion as shown on the lower half of the Figure B.13, where data with and without the blade are compared. This distortion produces a variation in the calculated flow coefficient and velocity coefficient as shown on Figure B.14. The data show that if inlet separation occurs in the wind tunnel at high nacelle angles of attack, thrust and air-flow calculated from coefficients obtained without distortion may be in error by several percent.

Actual distortion measured during the wind tunnel test for fan "B" (lift/cruise fan) and fan "C" (lift fan) are shown on Figures B.15 to B.25. Nose fan data is shown on Figure B.15 to B.19 for forward speeds up to 62 m/sec at high fan power settings. In most cases, the distortion level is well below the level produced during the FSC calibration. At 62 m/sec, distortion is comparable. Thus, at high nose fan power settings as would occur in conjunction with lift/cruise nacelle tilt angles near 90° , the nose fan distortion is within acceptable limits for thrust calculation within a few percent.

Figure B.19 shows nose distortion at a low power setting, statically and at 62 m/sec. Although no inlet separation is indicated, distortion again is high in the exit, producing exit velocity variations of about 3:1. In addition, exit static pressure variation was seen at low power settings, causing large changes in computed mass flow relative to the static case. These effects probably produce significant errors in predicted thrust.

Lift/cruise fan total pressure distortion is shown on Figure B.20 to B.25. For a tilt angle of 90° at 62 m/sec (120 knots), distortion greatly exceeds levels tested during the FSC calibration at all airflows. Although this condition is not part of the flight profile, it is representative of lower flight speeds and power settings resulting in a similar value C_j and inlet velocity ratio. At lower flight speeds and nacelle tilt angles of 50° and 0° , the distortion is acceptable from the standpoint of thrust calculation accuracy.

Primary nozzle total pressure surveys using a translating probe were integrated to produce average total pressure from which flow and thrust were calculated. The survey results were related to a manifolded rake. Thrust and flow coefficients were then calculated for use with manifold pressure data. The results for primary nozzles "A" and "B" are tabulated below:

Nozzle	A	B	Remarks
C_V	.950	.968	
C_D	.950	.968	(based on ambient static)
C_D	1.40	1.43	(based on nozzle internal static pressure)

Fan nozzle total pressure profiles measured by six rakes with 4 probes each were fairly flat. An example for fan "B" at a fan total pressure ratio near 1.14 is shown as the "without blade" case in Figure B.13.

The inlets of fans "A" and "B" were calibrated for airflow. Figure B.26 presents an inlet flow coefficient defined as the ratio of actual airflow to theoretical airflow which is based on measured inlet static pressure, temperature and flow area. The coefficient was approximately 0.905 for both inlets.

For reduction of the static ground test data, thrust was related to fan RPM due to difficulties with pressure instrumentation. The relationship of thrust and fan RPM used is shown in Figure B.27. It was determined from an in-place calibration using the model internal balance to measure thrust. The fans were run individually with the model fan out of ground effects (ground height - 1.8 meters (72 inches) and the lift/cruise fans tilted to 90°. The calibration was not extended to low RPM in that the reliability of the indicated RPM deteriorates considerably at low RPM. The static tests were all conducted at or near 28,000 RPM.

During the wind tunnel testing the difficulties with the fan exit pressure instrumentation were rectified and an in-place calibration was conducted to relate thrust to fan exit instrumentation as had been done in the flight simulation chamber. The in-place calibration indicated that the thrust measured was lower than the calculated thrust based upon the flight simulation chamber coefficients. Fans "A" and "B" were consistently 2% low while fan "C" was 4% low. Thus the flight simulation chamber calibration results were adjusted accordingly for calculating thrust during the wind tunnel testing.

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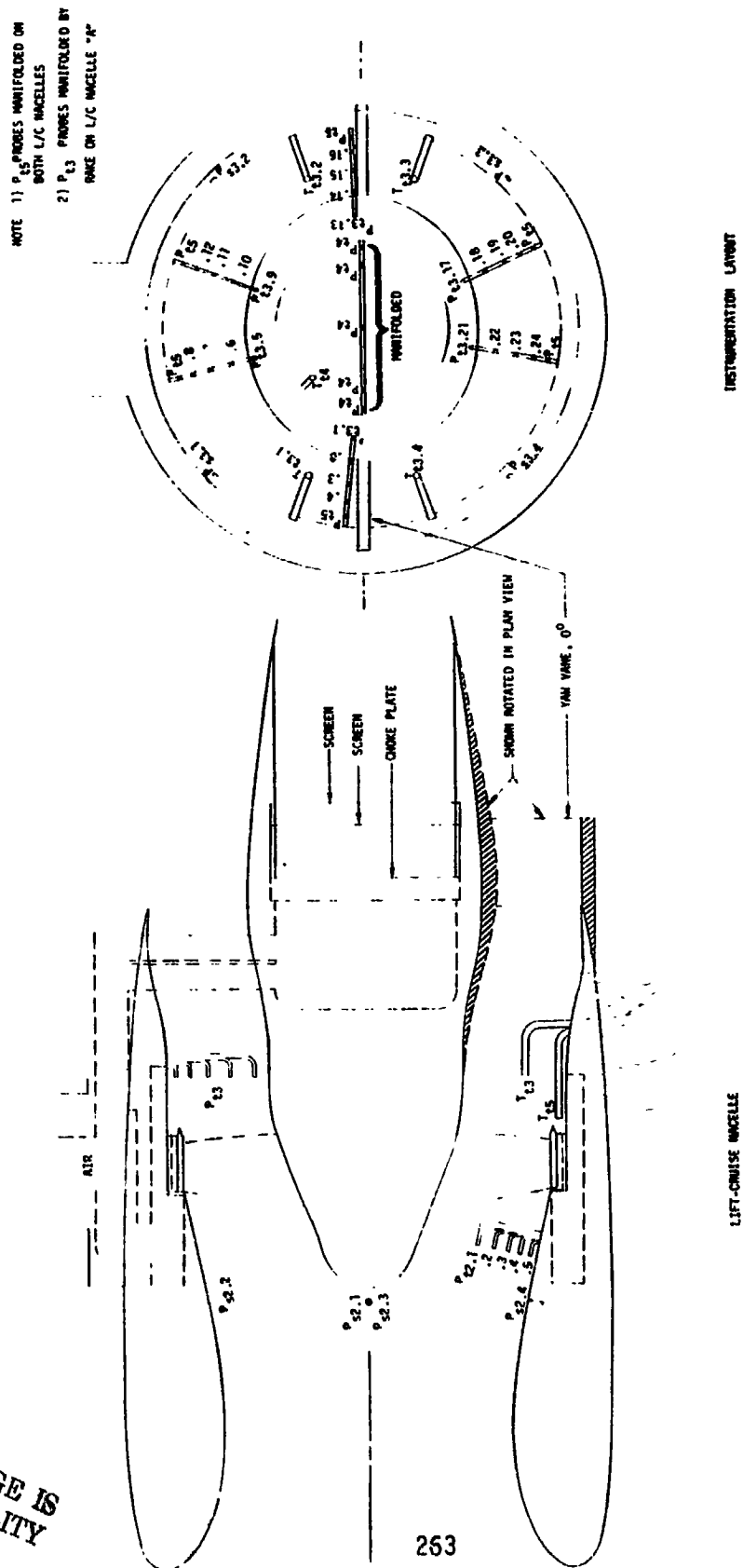
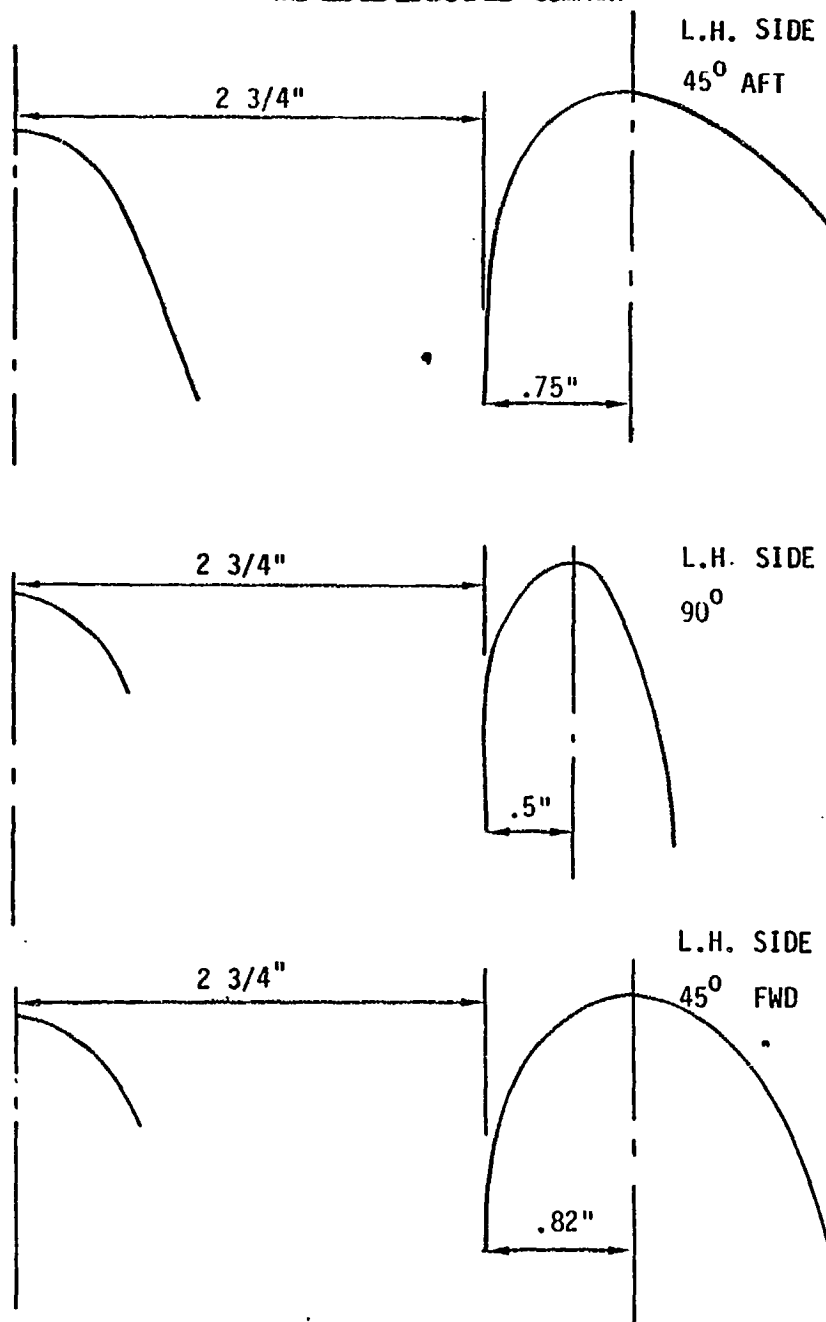


FIGURE B.1 LIFT-CRUISE FAN NACELLE MODEL

LIFT/CRUISE INLET COORDINATES					
(SEE FIG B.1 FOR INLET LINES)					
OUTSIDE			INSIDE		
	STATION INCHES	RADIUS INCHES		STATION INCHES	RADIUS INCHES
HILITE →	0	3.0818		0	3.818
	.0068	3.1465			
	.0223	3.1969		.0083	3.0022
	.0942	3.3038		.0271	2.9396
	.1457	3.3453		.0520	2.8837
	.2228	3.3897		.1145	2.7893
	.3000	3.4214		.1770	2.7173
	.3771	3.4461		.2707	2.6328
	.4542	3.4651		.3643	2.5643
	.6084	3.4951		.4581	2.5058
	.7627	3.5195		.5518	2.4592
	.9169	3.5407		.7391	2.3720
	1.0712	3.5594		.9266	2.3070
	1.2255	3.5760		1.1139	2.2563
	1.3797	3.5907		1.0314	2.2176
	1.5340	3.6030		1.4889	2.1894
	1.6882	3.6154		1.6762	2.1709
	1.8425	3.6256		1.8637	2.1613
	1.9967	3.6344	THROAT →	2.0510	2.1599
	2.1510	3.6419		2.2385	2.1606
	2.3052	3.6486		2.4258	2.1676
	2.6137	3.6576		2.6133	2.1816
	2.9222	3.6625		2.8006	2.2014
MAX	3.1239	3.6634		3.1754	2.2265
				3.5502	2.2887
				3.7954	2.3614

FIGURE B.2 LIFT/CRUISE NACELLE INLET COORDINATES

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CONTRACTION RATIOS (AHI/ATH)

CENTER, FWD	AHI/ATH = 1.76 (AS DRAWN IN FIGURE B.4)
L.H. SIDE 45° FWD	" = 1.69
" 90°	" = 1.40
" 45° AFT	" = 1.62

FIGURE B.3 MODEL NOSE FAN INLET CONTOURS

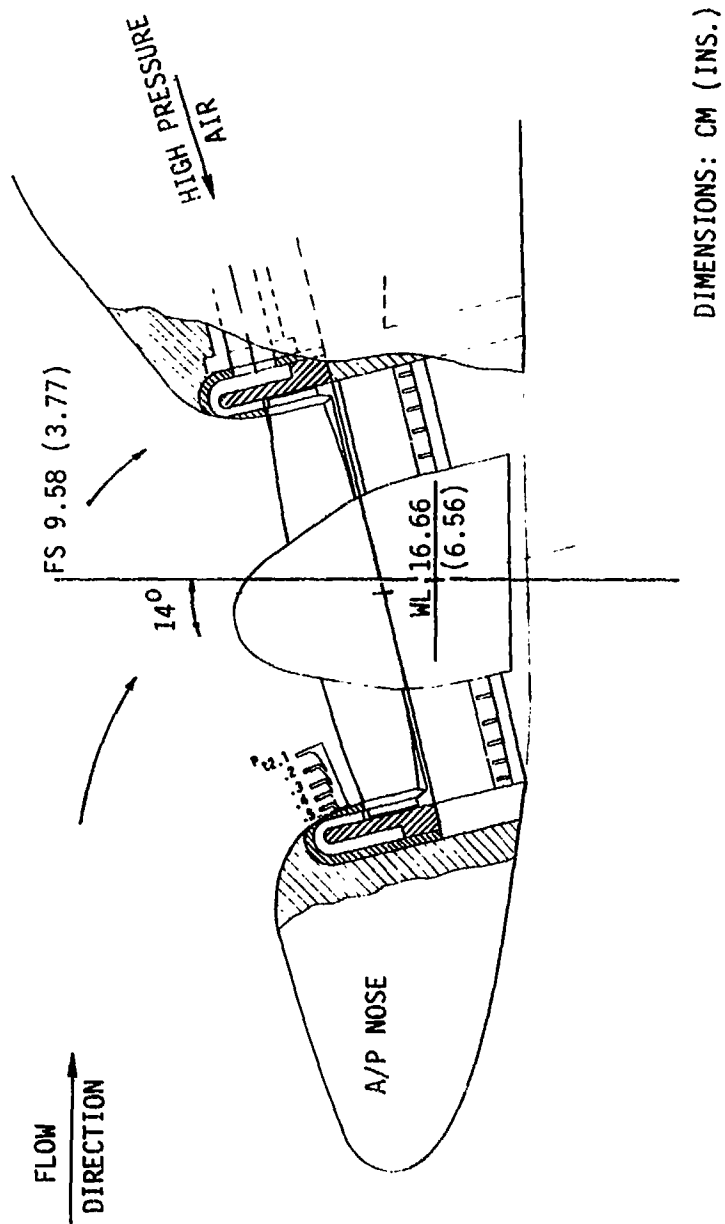


FIGURE B.4 NOSE FAN ASSEMBLY

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FIGURE B.5 - INSTRUMENTATION OF LIFT/CRUISE FAN NOZZLES

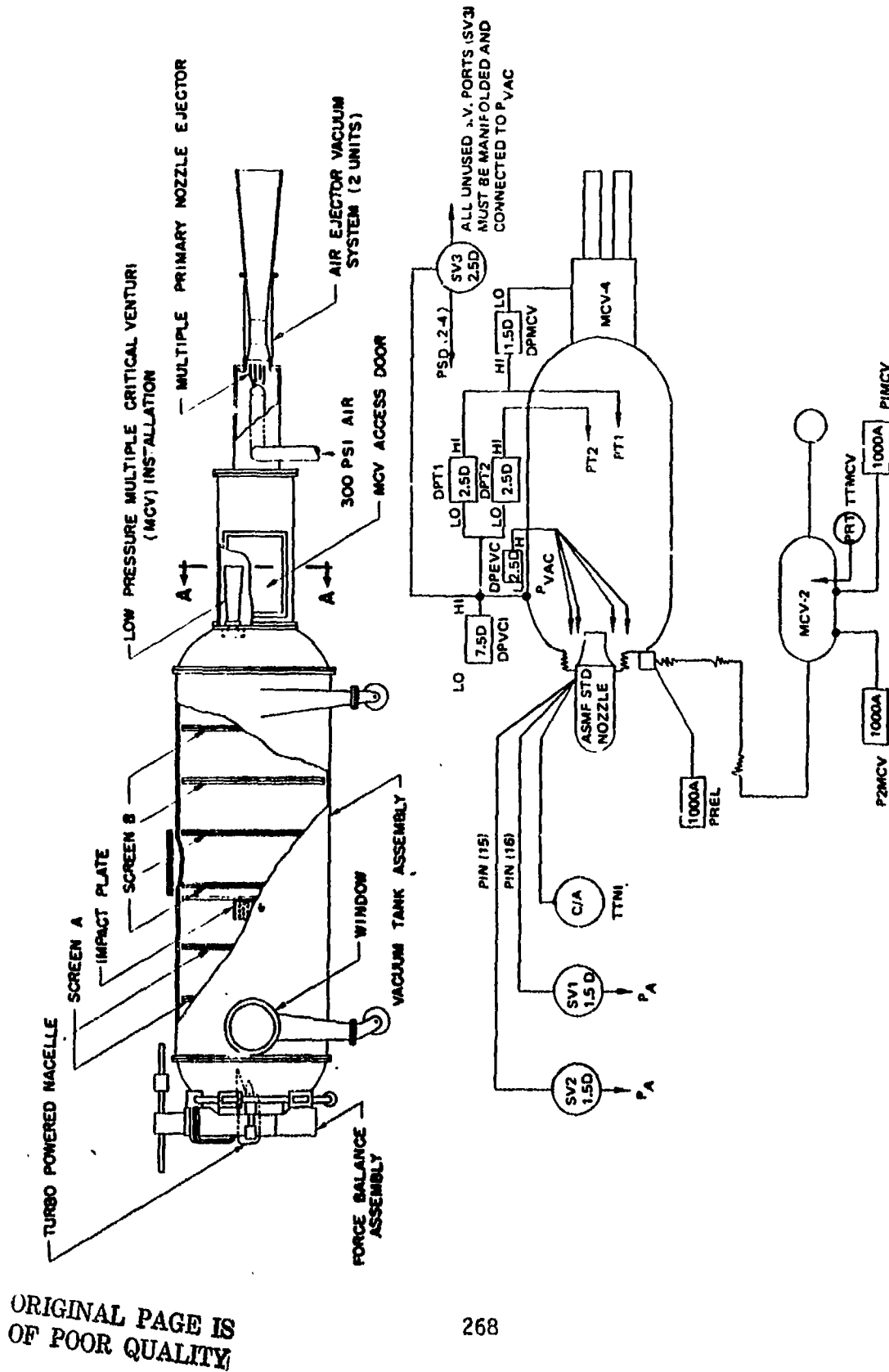
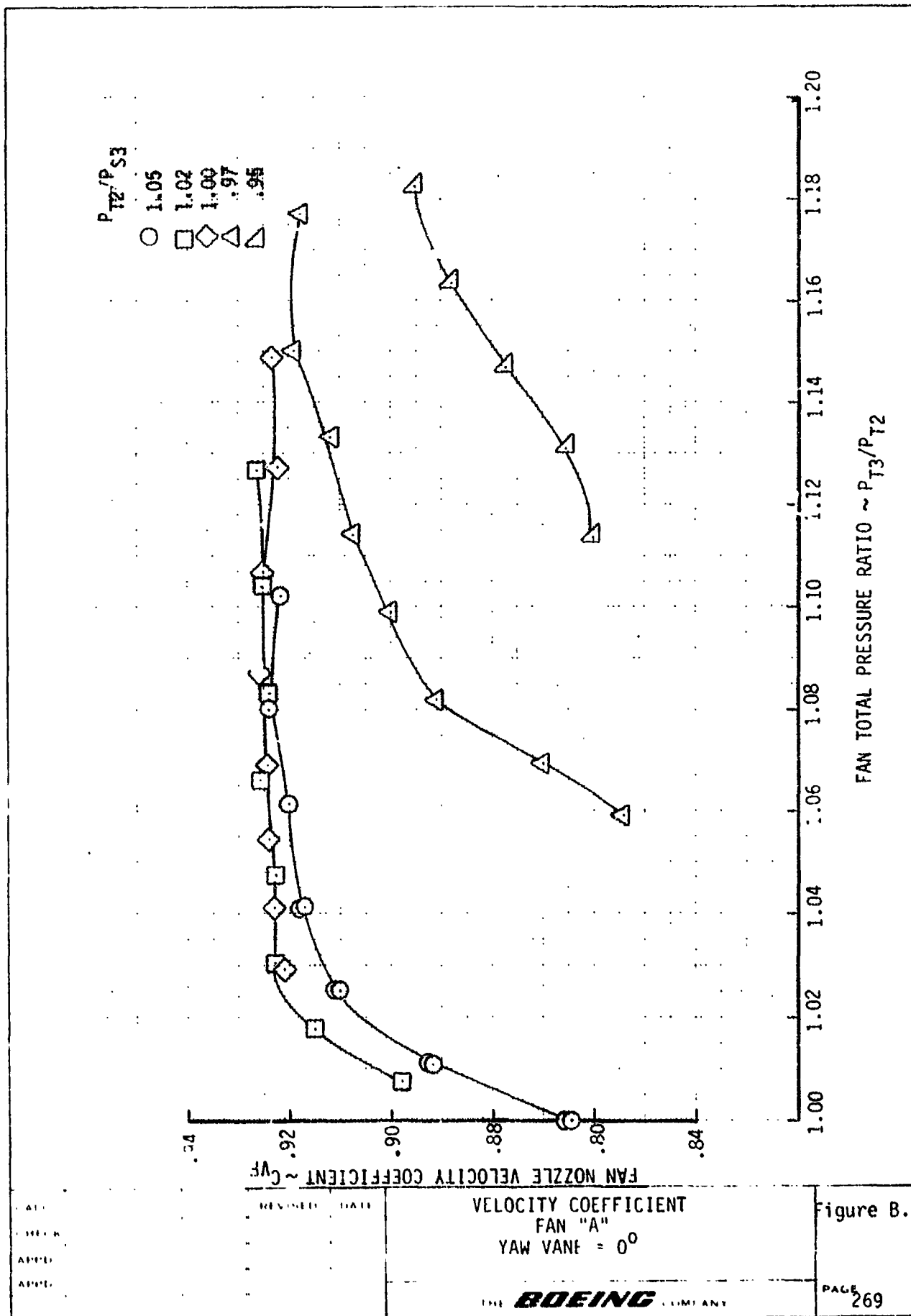


FIGURE B.6 FLIGHT SIMULATION CHAMBER

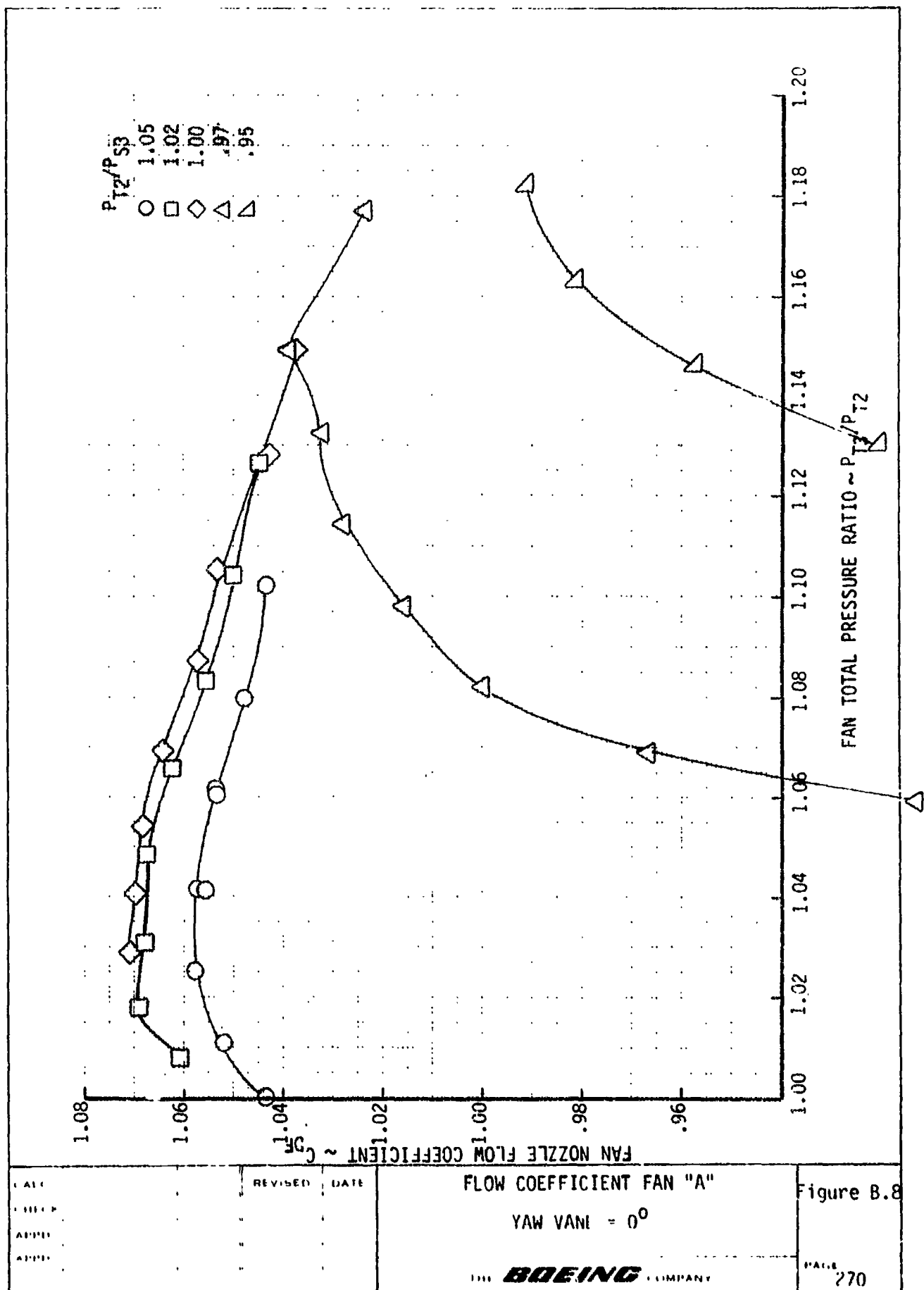
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FAN NOZZLE VELOCITY COEFFICIENT $\sim C_{VE}$

VELOCITY COEFFICIENT
 FAN "A"
 YAW VANE = 0°

Figure B.7



FLOW COEFFICIENT FAN "A"

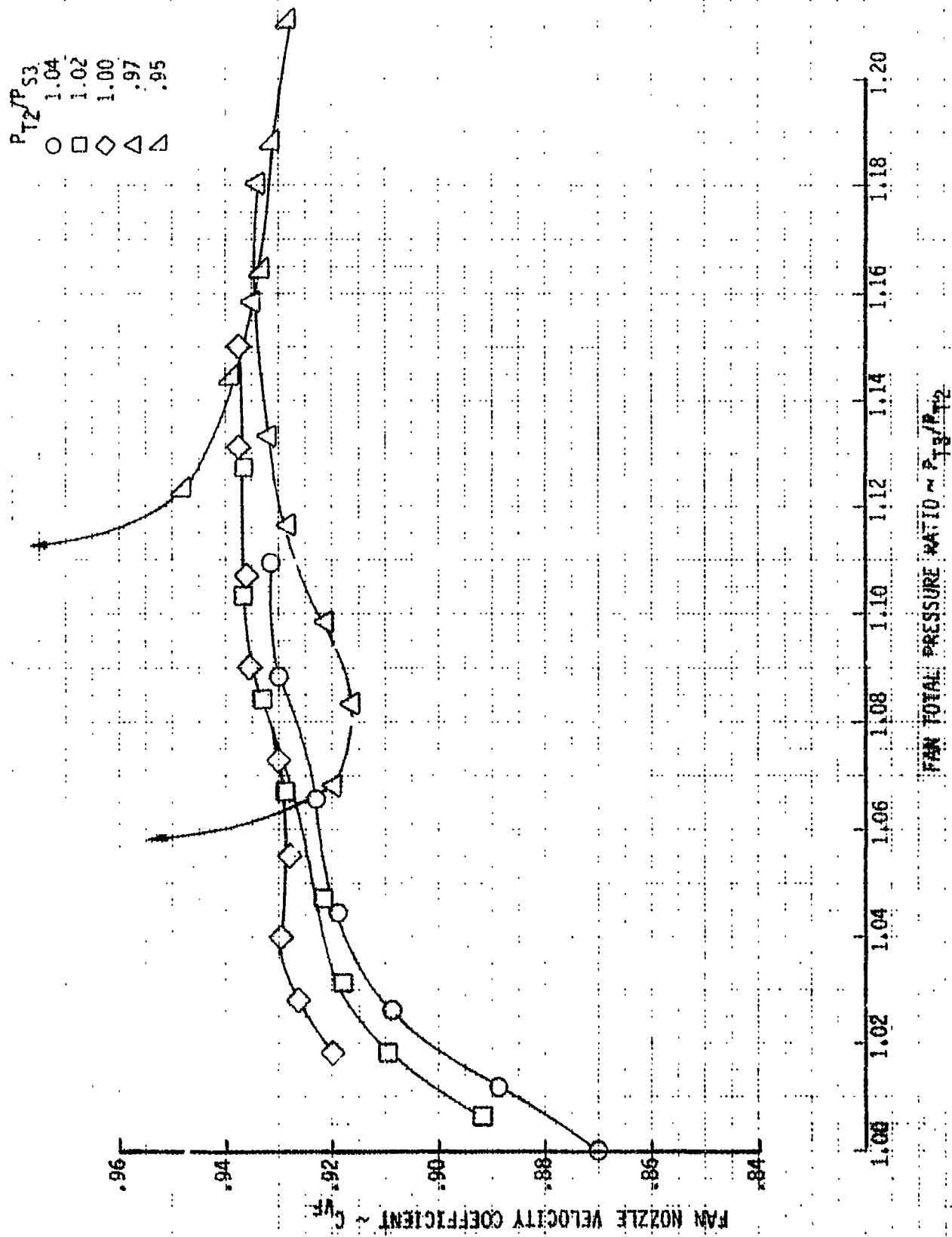
YAW VANE = 0°

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Figure B.8

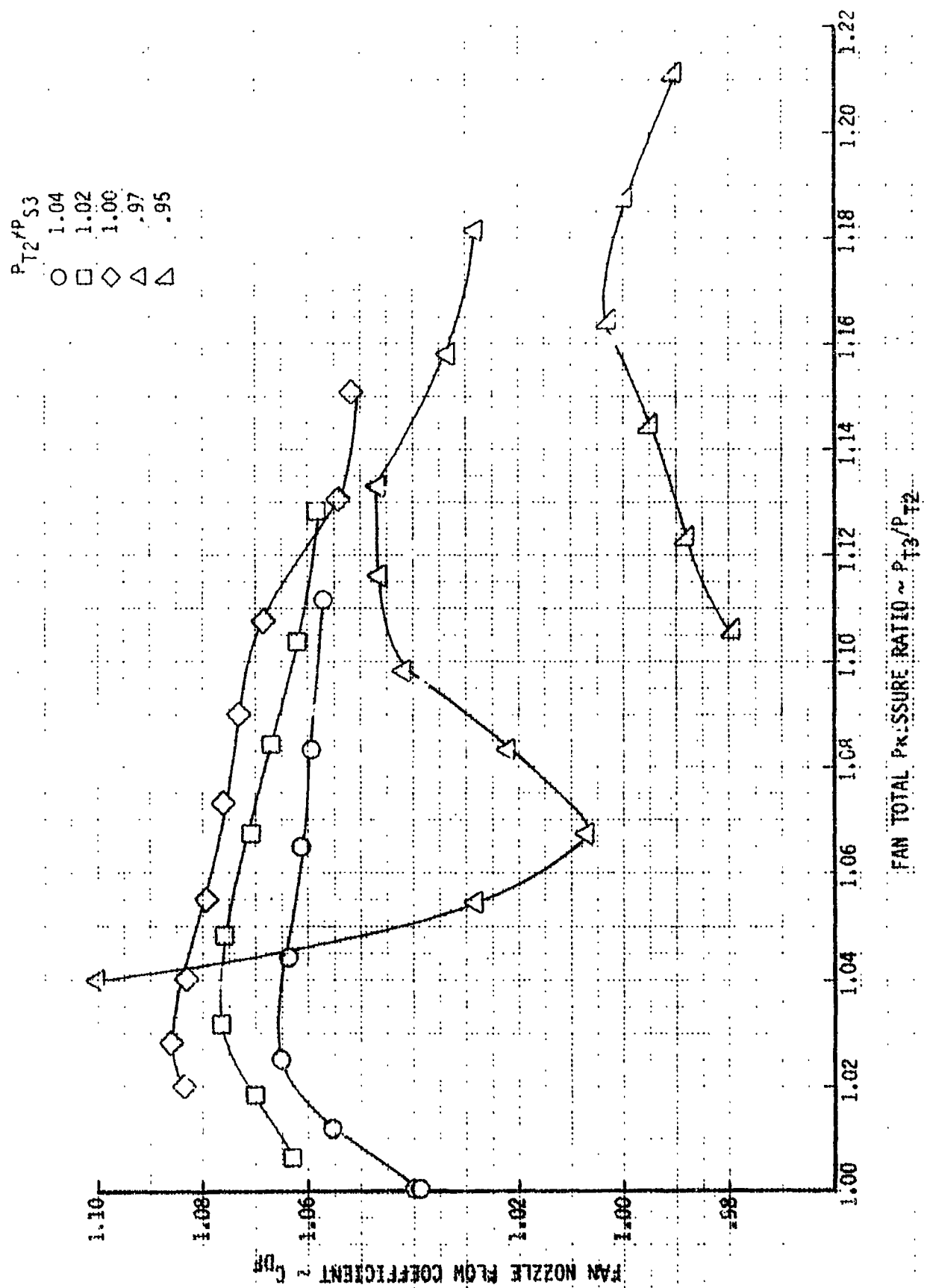
PAGE 270

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VELOCITY COEFFICIENT - FAN "B"
 YAW VANES = 0°

Figure B.9

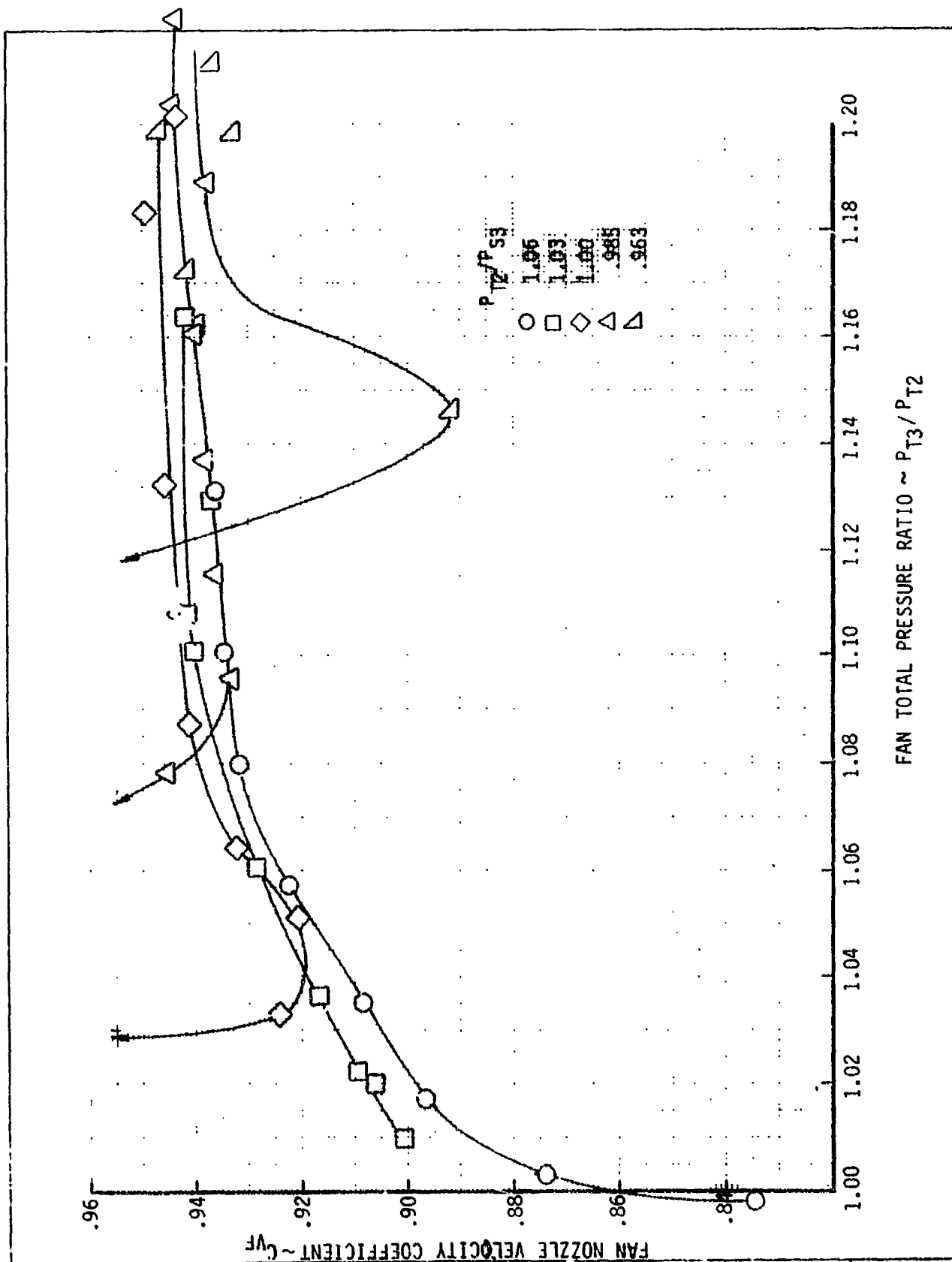


FLOW COEFFICIENT - FAN "B"

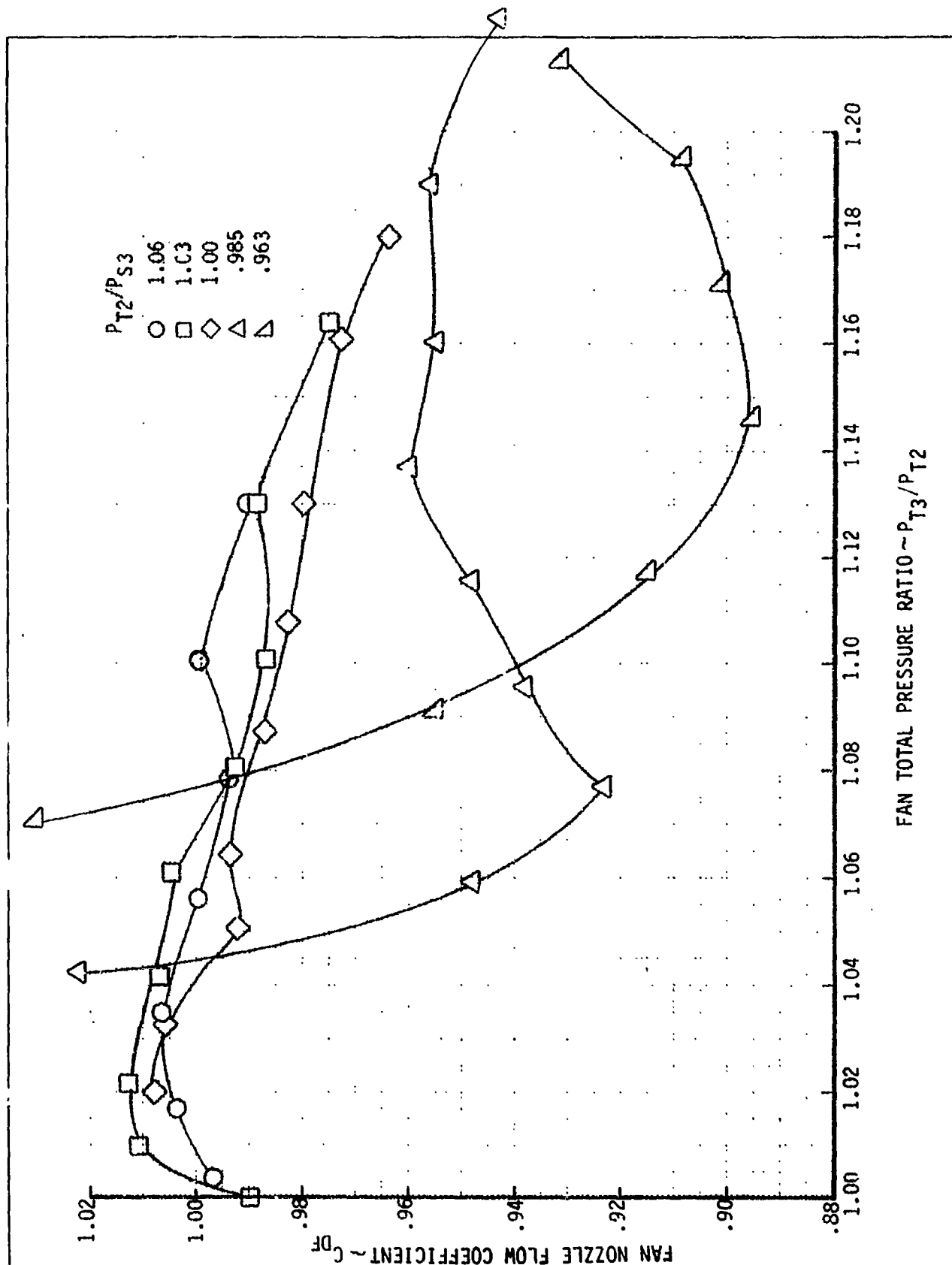
YAW VANES = 0°

THE BRIDGES COMPANY

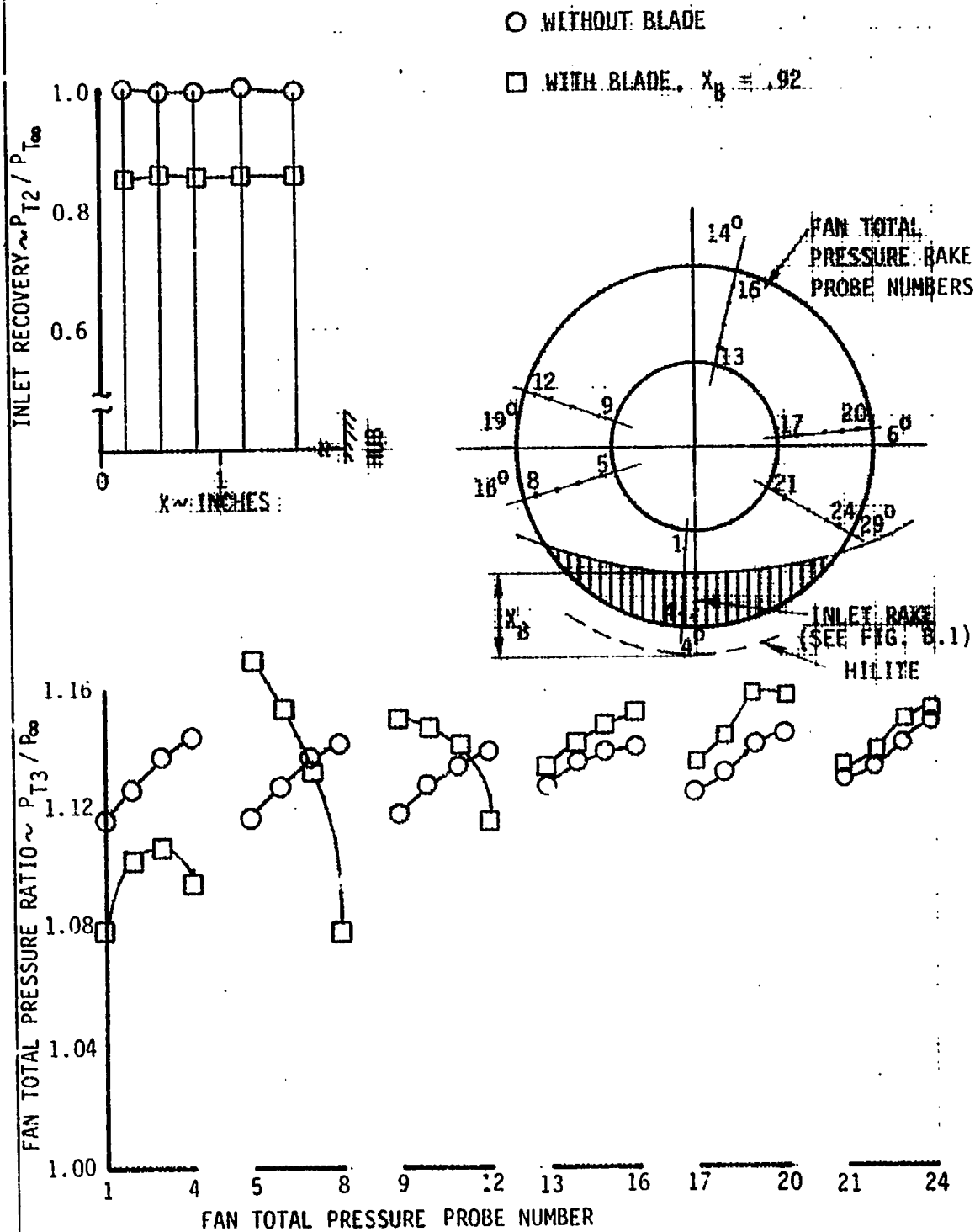
Figure B.10



FAN TOTAL PRESSURE RATIO $\sim P_{T3}/P_{T2}$



CALC CHECK APPD APPD	REVISED DATE	FLOW COEFFICIENT - FAN "C" YAW VANES = 0°	Figure B.12
		THE BOEING COMPANY	PAGE 274



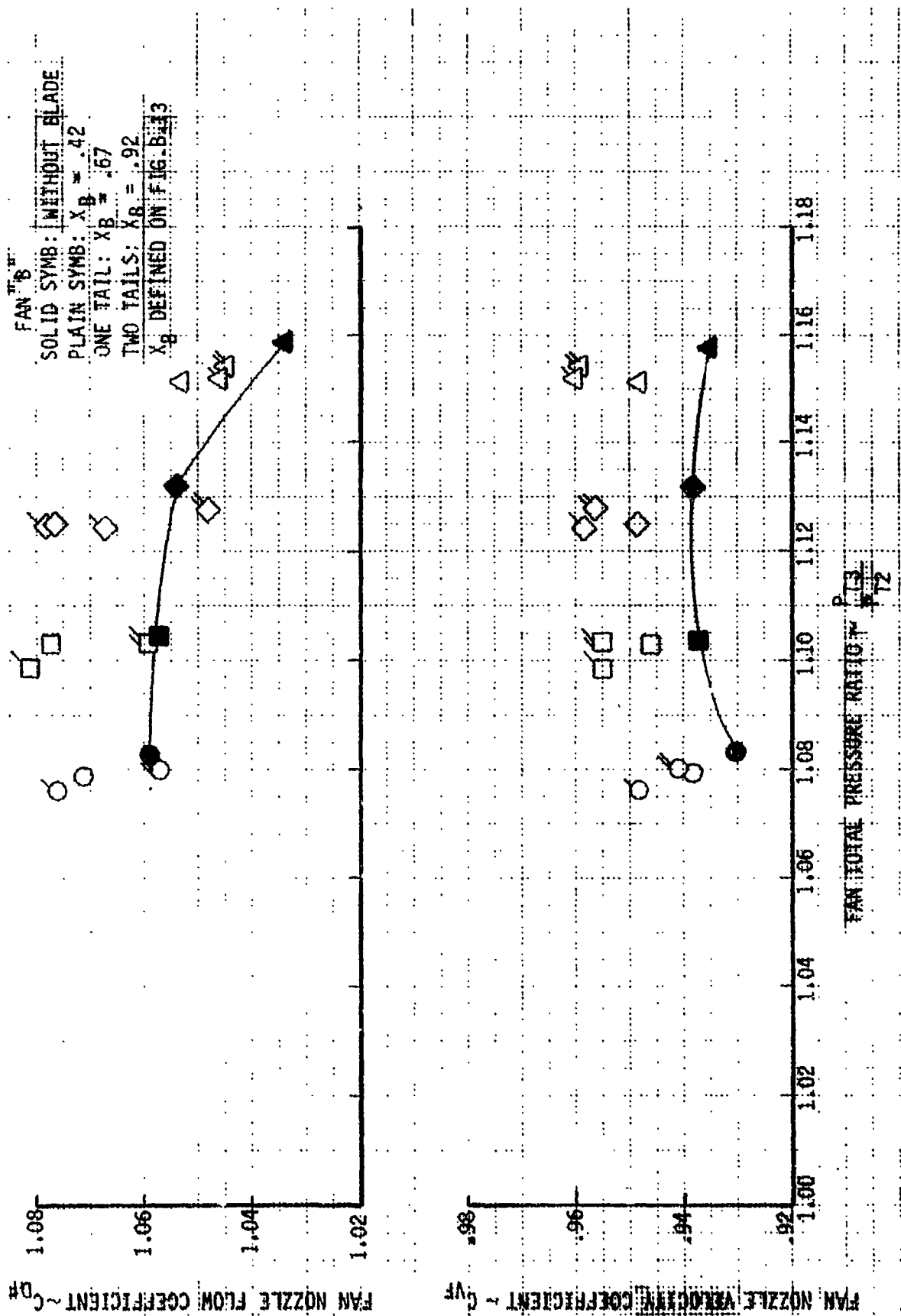
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APPD		

EFFECT OF INLET DISTORTION
ON FAN NOZZLE DISTORTION - FAN B

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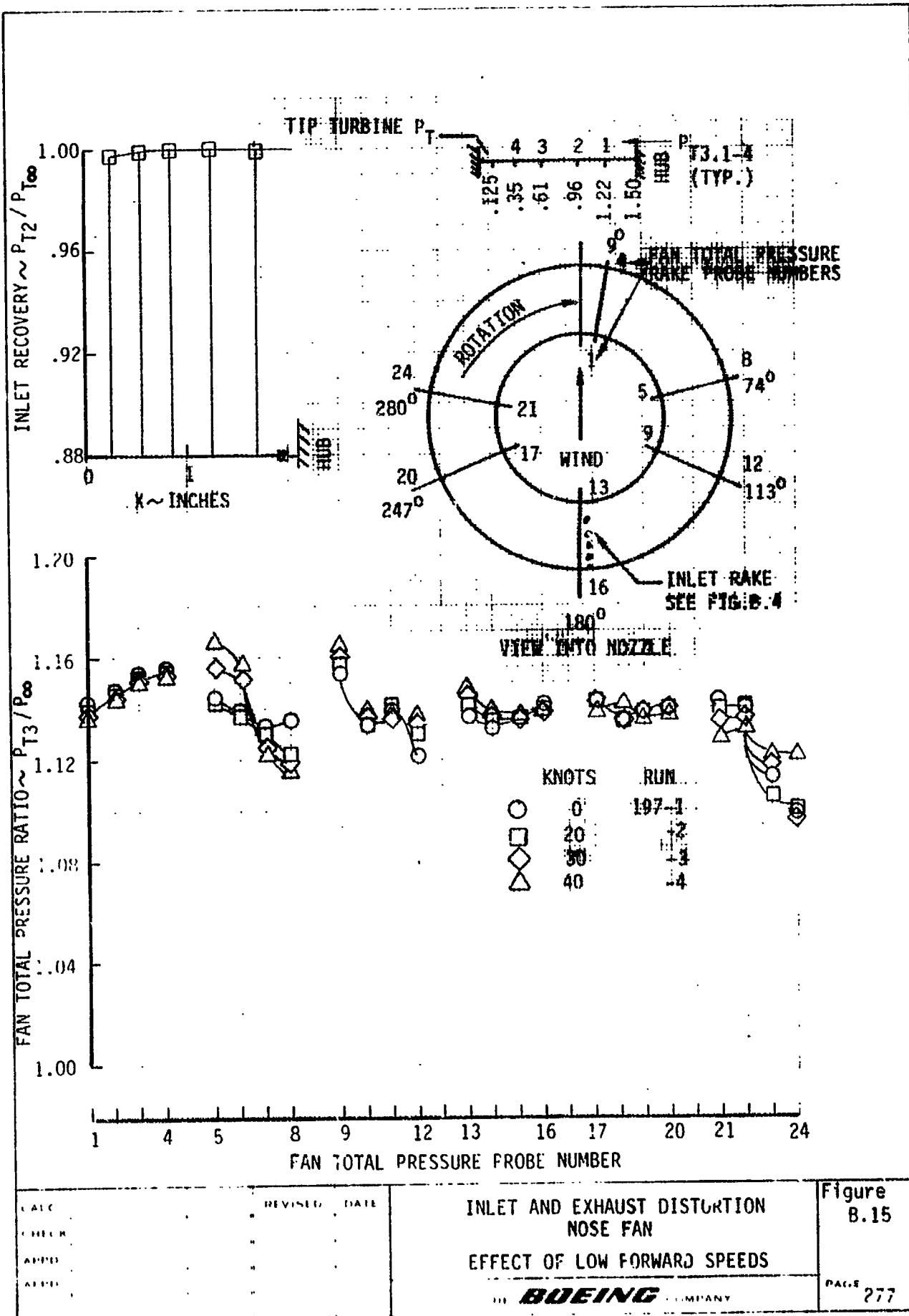
FIG. B.13

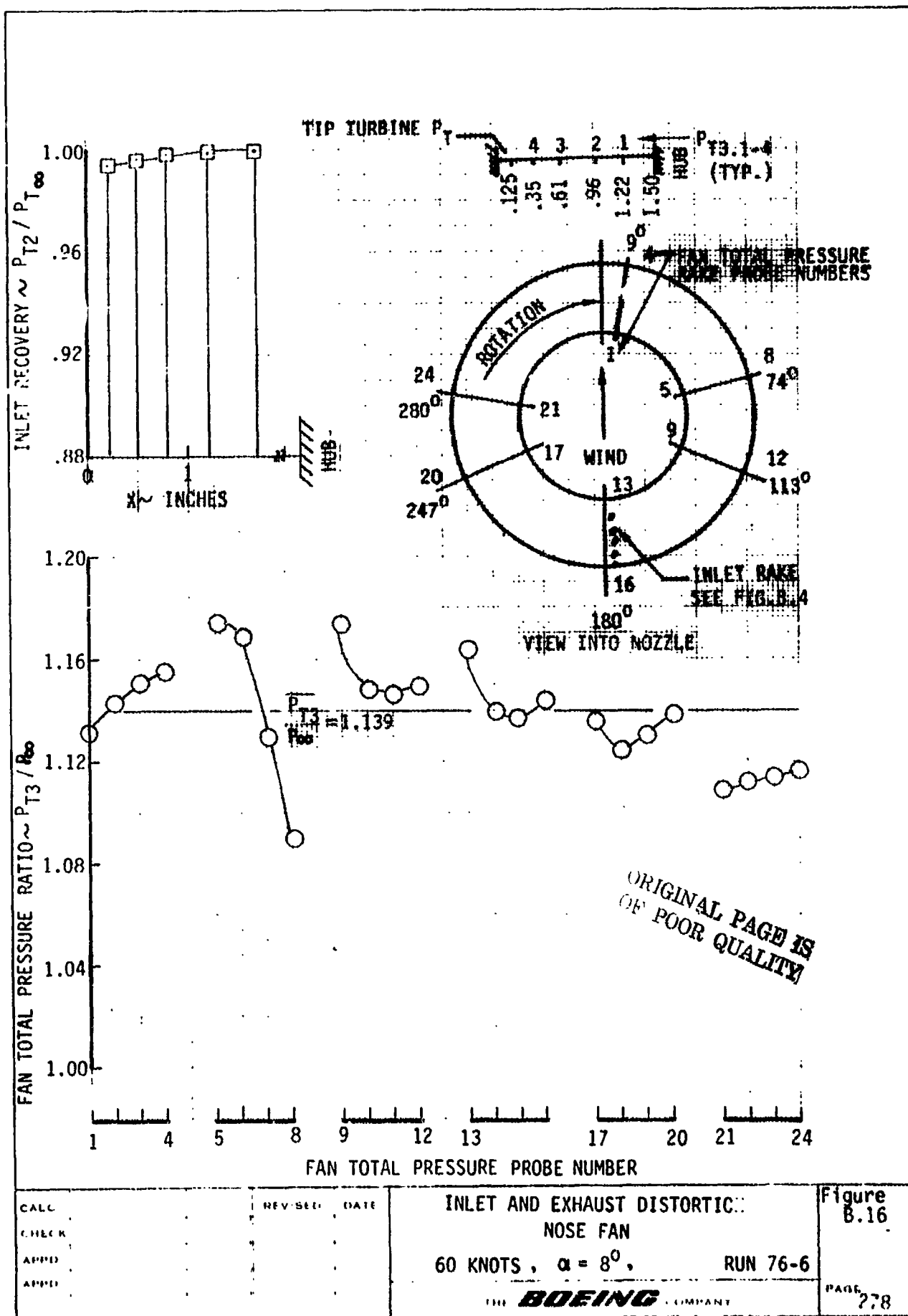
PAGE 275



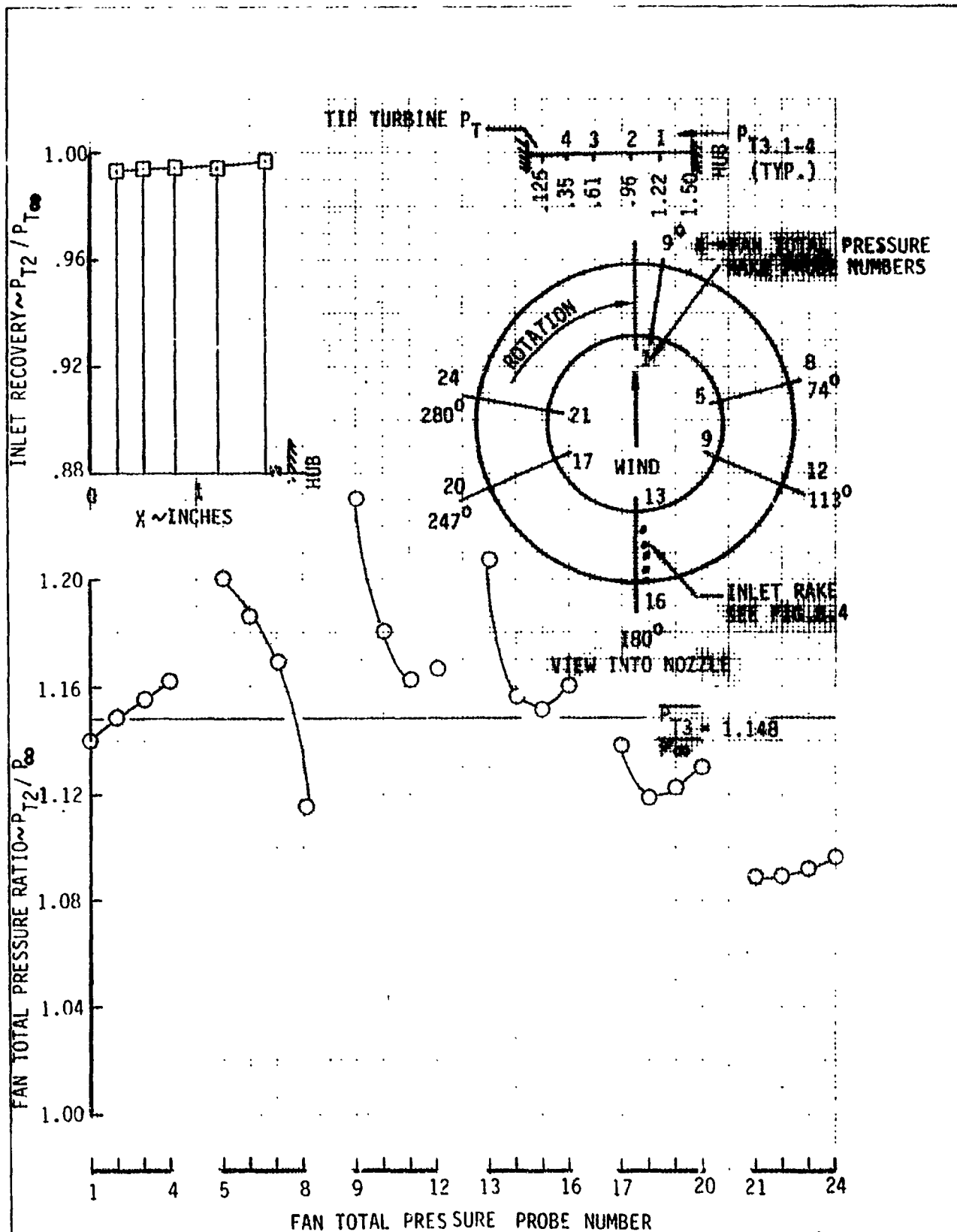
EFFECT OF FAN EXIT TOTAL
PRESSURE DISTRIBUTION
ON NOZZLE COEFFICIENTS

Figure B.14





CALL	REV	DATE	INLET AND EXHAUST DISTORTION	Figure B.16
CHECK			NOSE FAN	
APPRO			60 KNOTS, $\alpha = 8^\circ$, RUN 76-6	
APPRO			THE BOEING COMPANY	PAGE 278



INLET AND EXHAUST DISTORTION
NOSE FAN

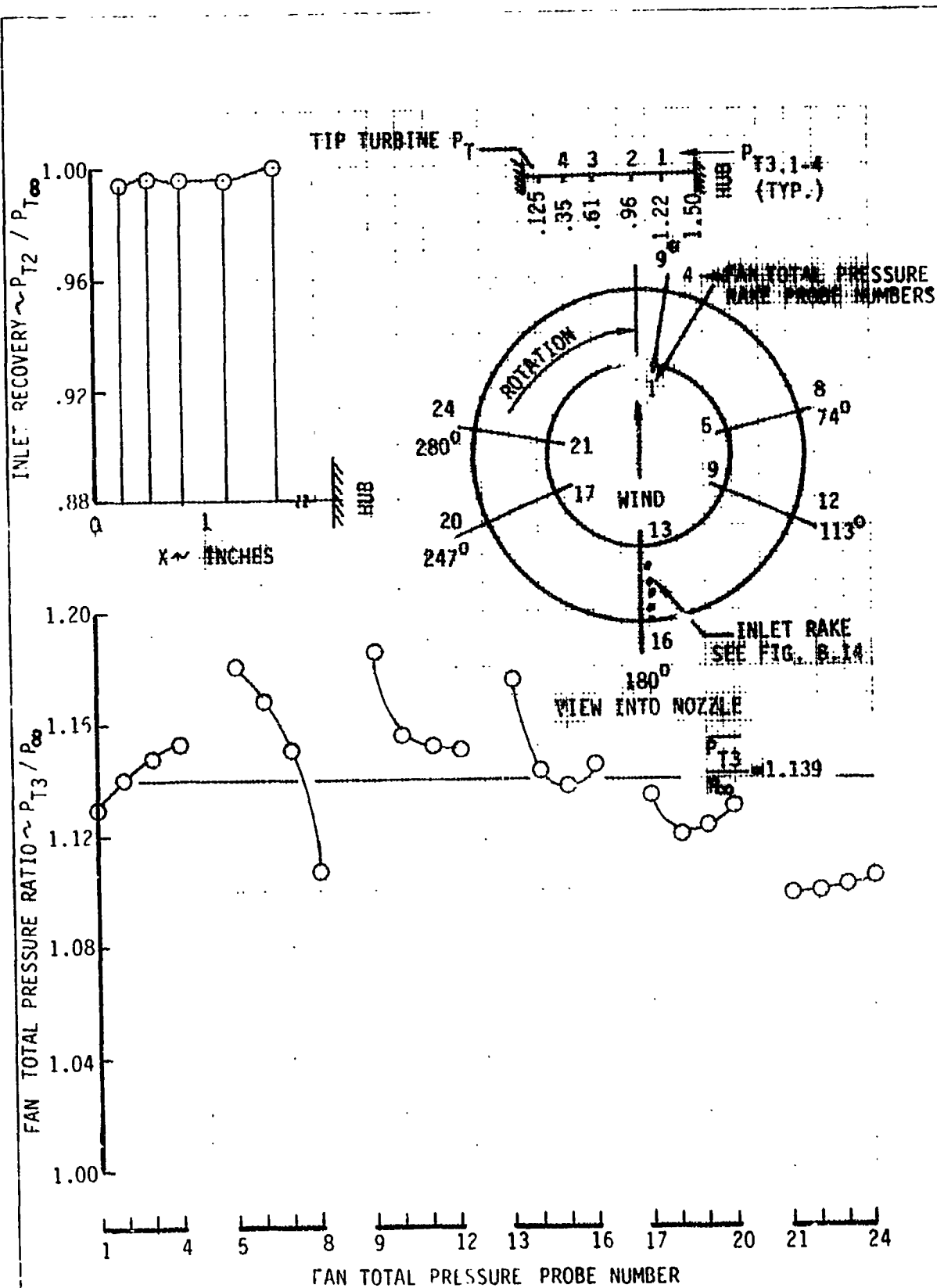
120 KNOTS $\alpha = 8^\circ$

RUN 69-6

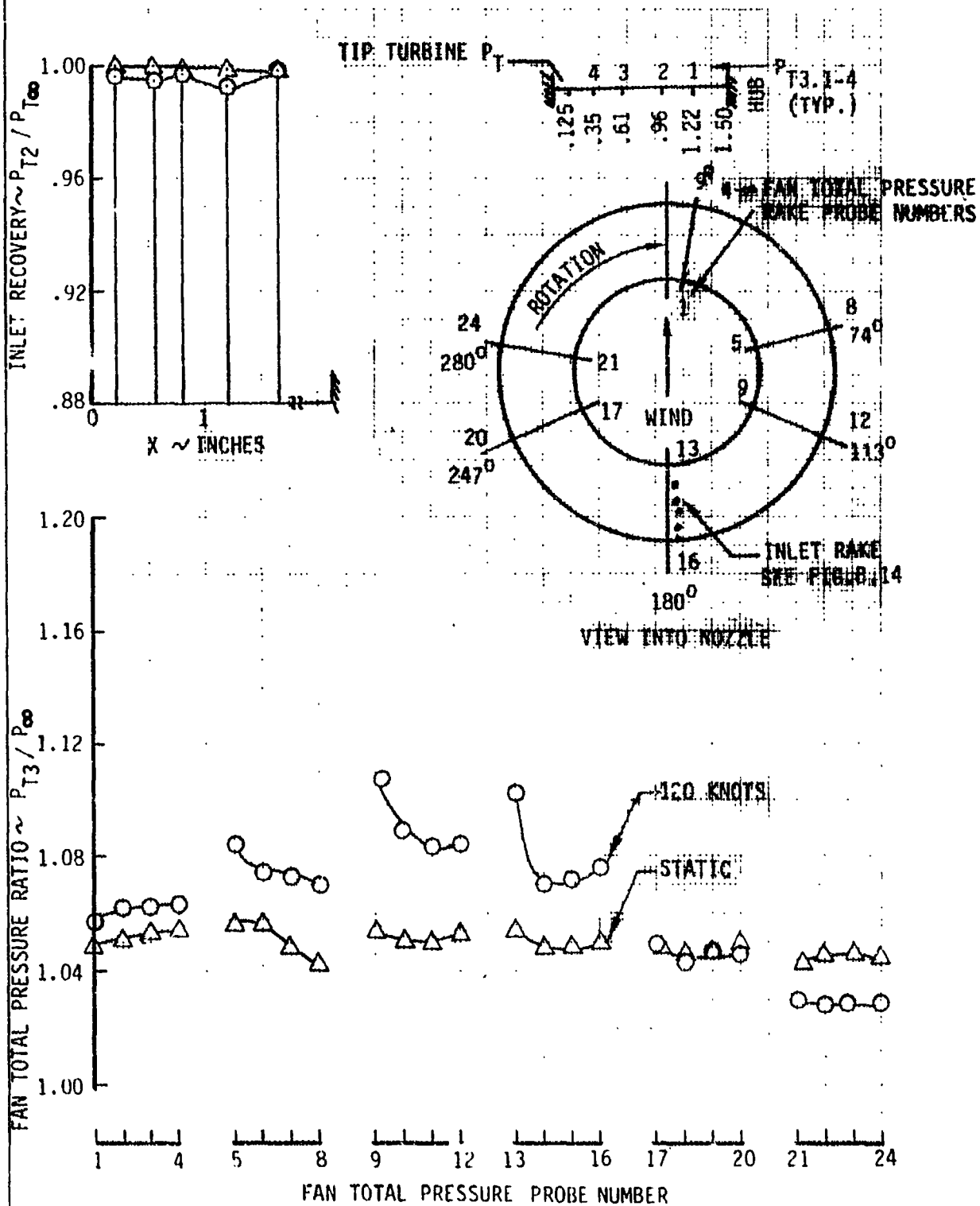
Figure
B.17

PAGE
279

BOEING



DATE CHECK APPD APPD	REV SET DATE	INLET AND EXHAUST DISTORTION NOSE FAN 80 KNOTS, $\alpha = 8^\circ$ RUN 71-13 THE BOEING COMPANY	Figure 8.18 PAGE 2P0
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CALL
CHECK
APPD
APPD

REVISED DATE

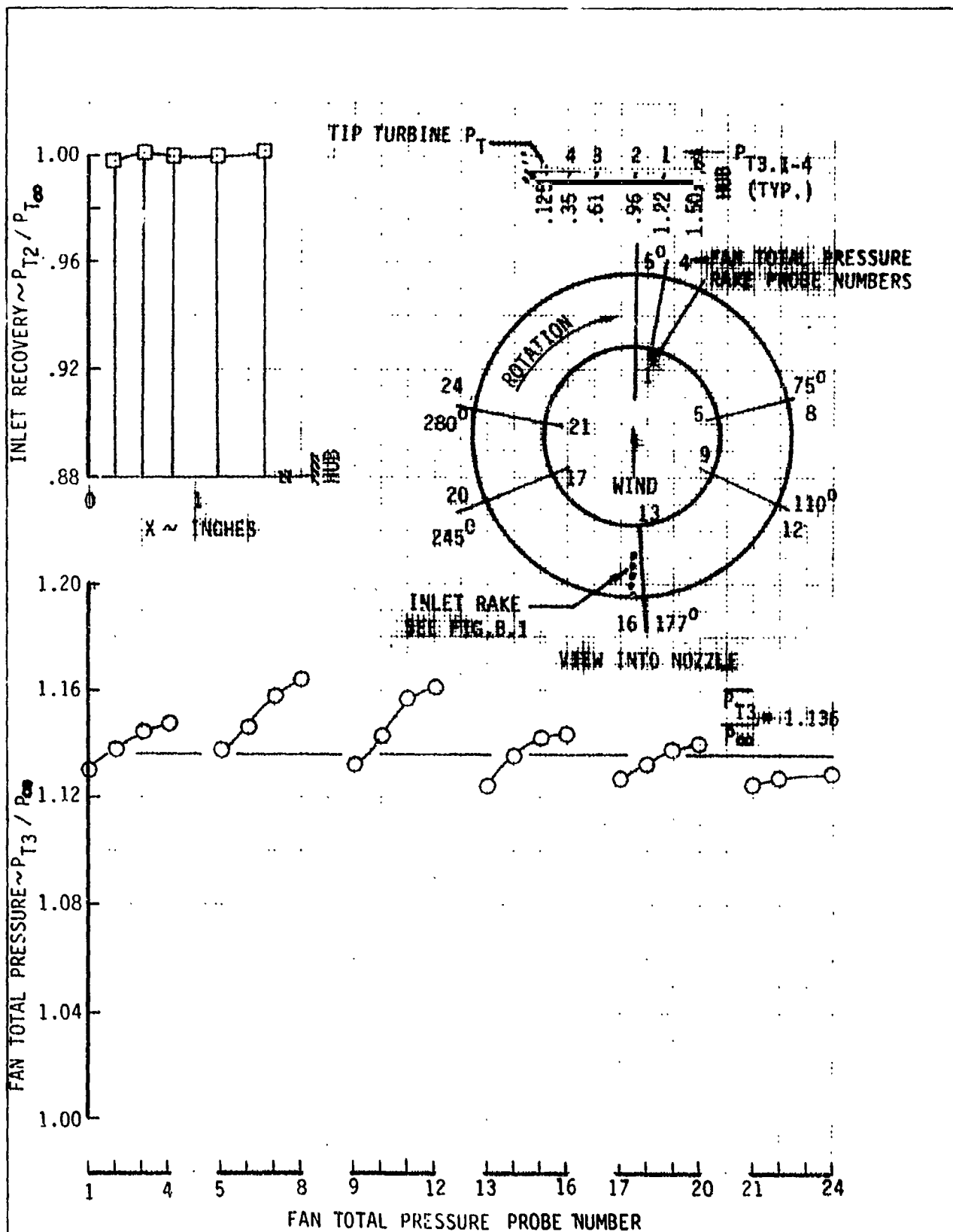
INLET AND EXHAUST DISTORTION
NOSE FAN
EFFECT OF SPEED AT LOW THRUST
 $\alpha = 12^\circ$

RUN 229-1,-9

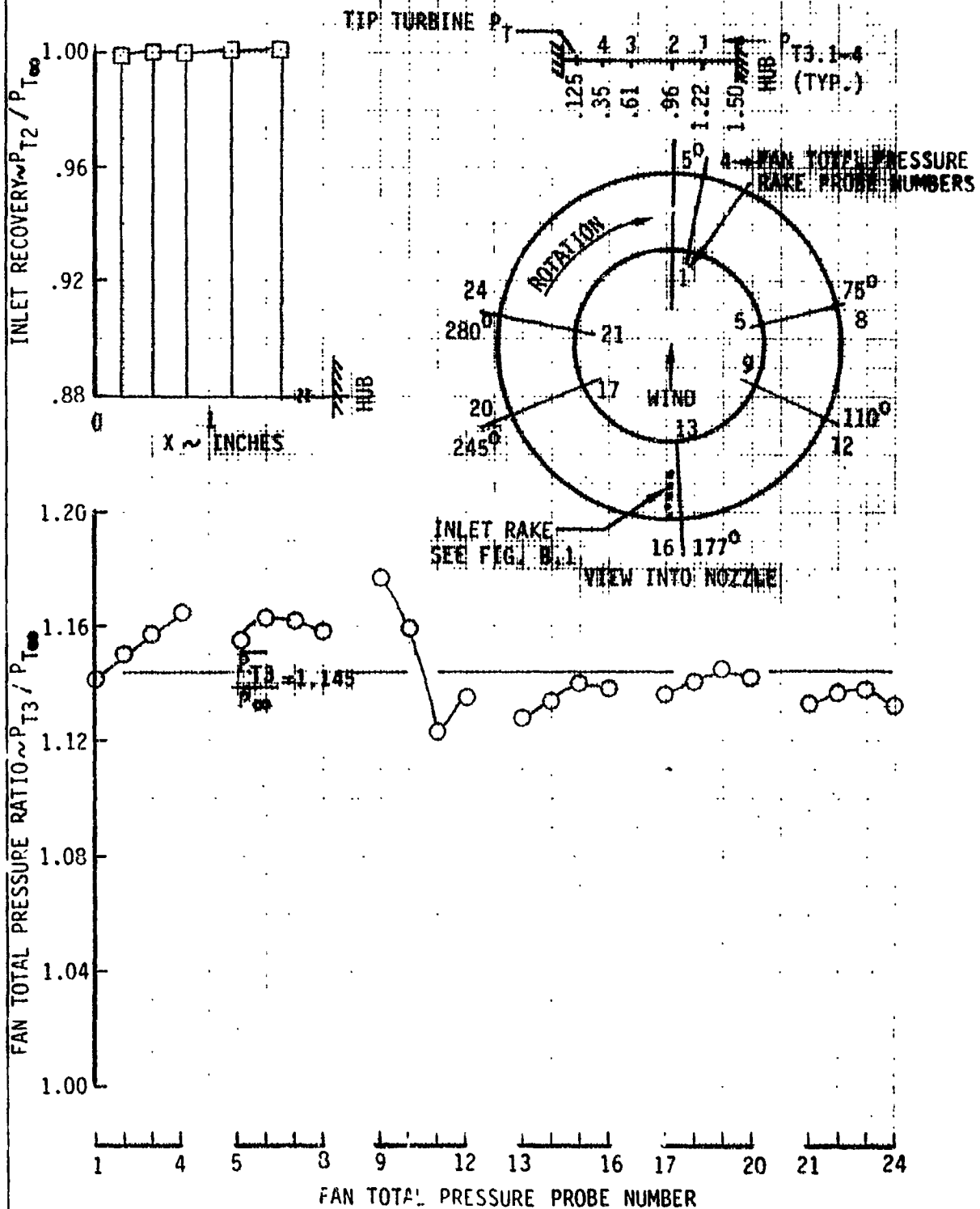
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Figure
B.19

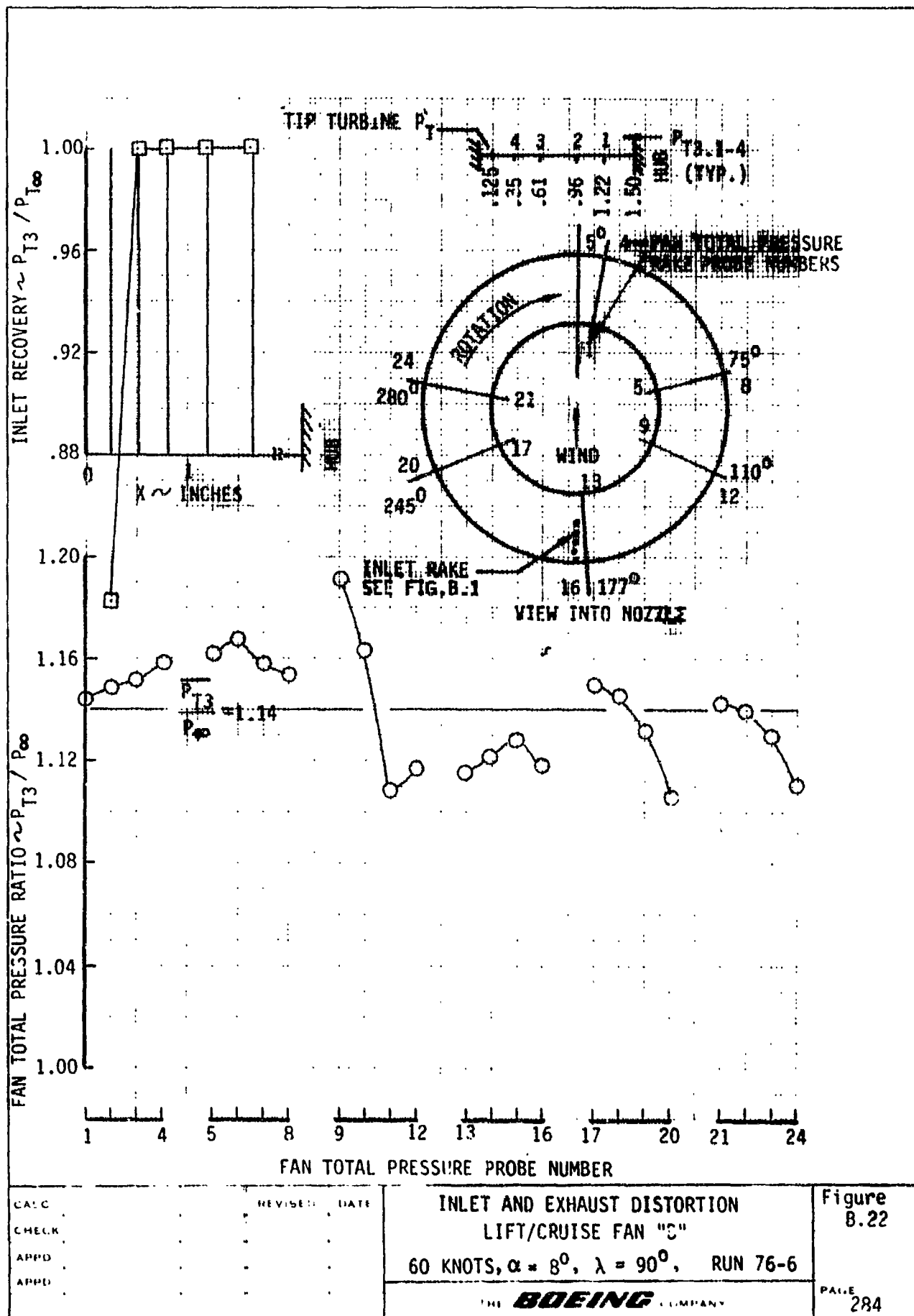
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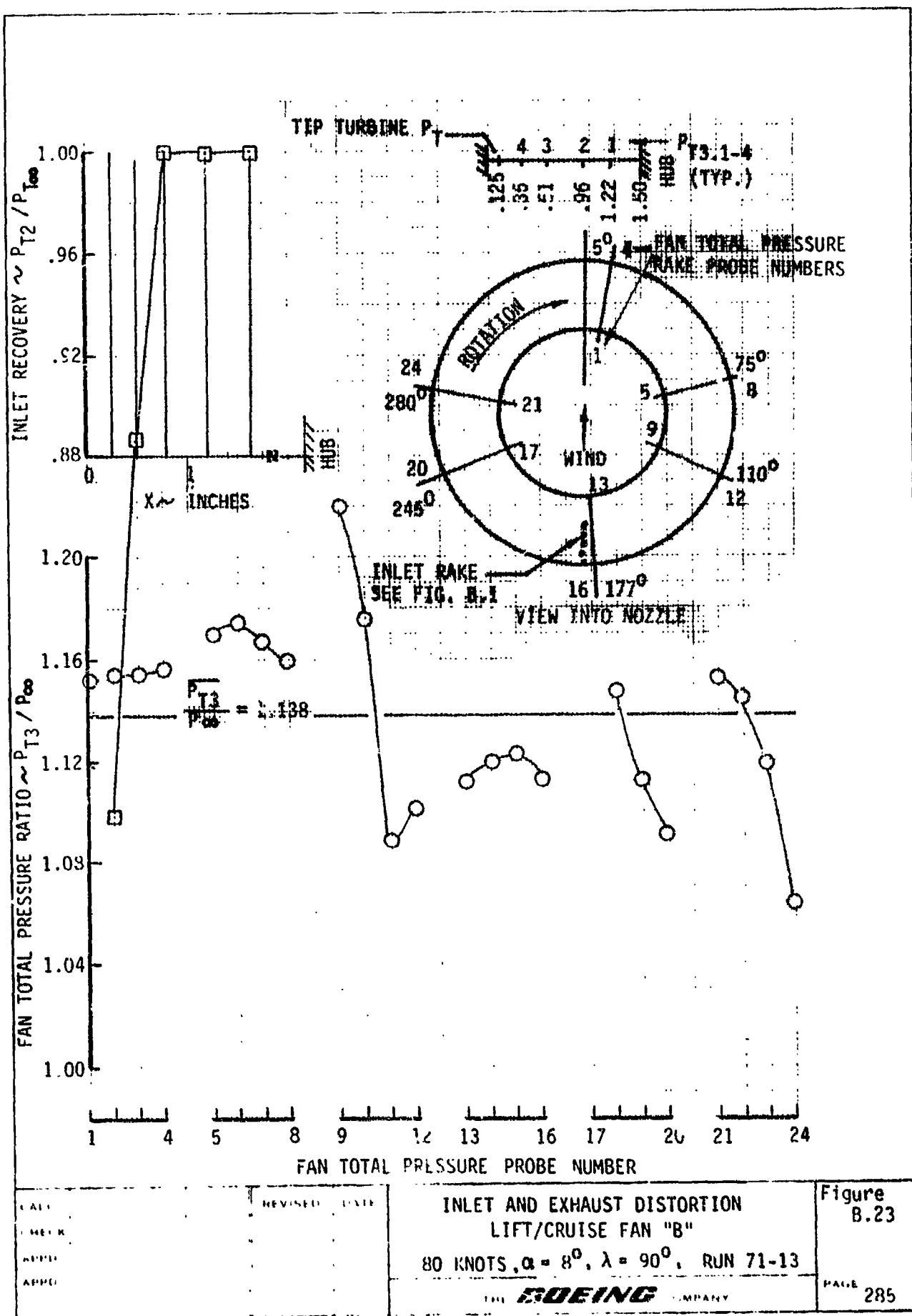


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<div> <div>THE</div> <div>BOEING</div> <div>COMPANY</div> </div>			<div> <div>PAGE</div> <div>282</div> </div>

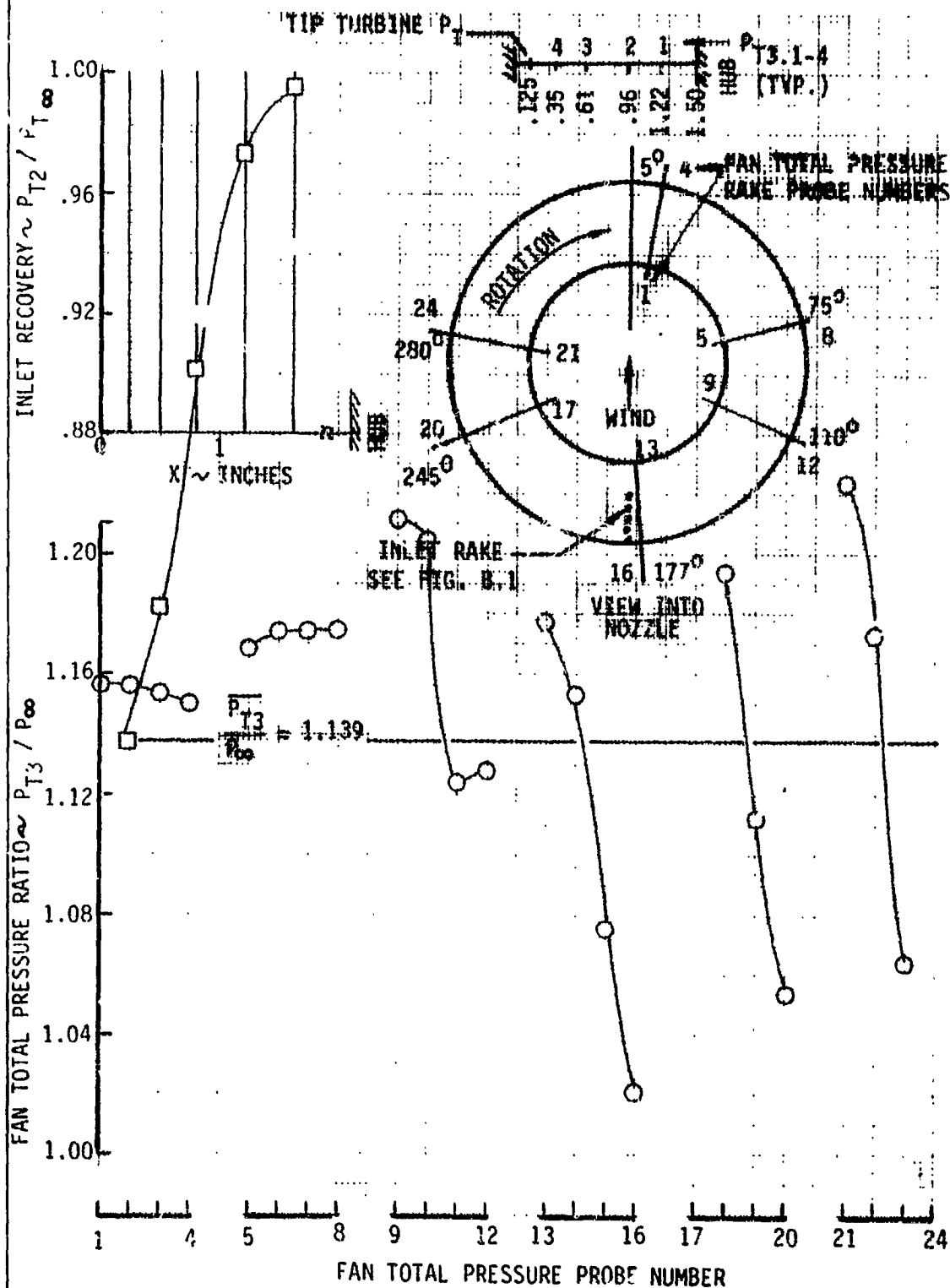


CALC CHECK APPD APPD	REVISED DATE	INLET AND EXHAUST DISTORTION LIFT/CRUISE FAN "B" 40 KNOTS, $\alpha = 8^\circ$, $\lambda = 90^\circ$, RUN 76-10 THE BOEING COMPANY	Figure B.21 PAGE 283
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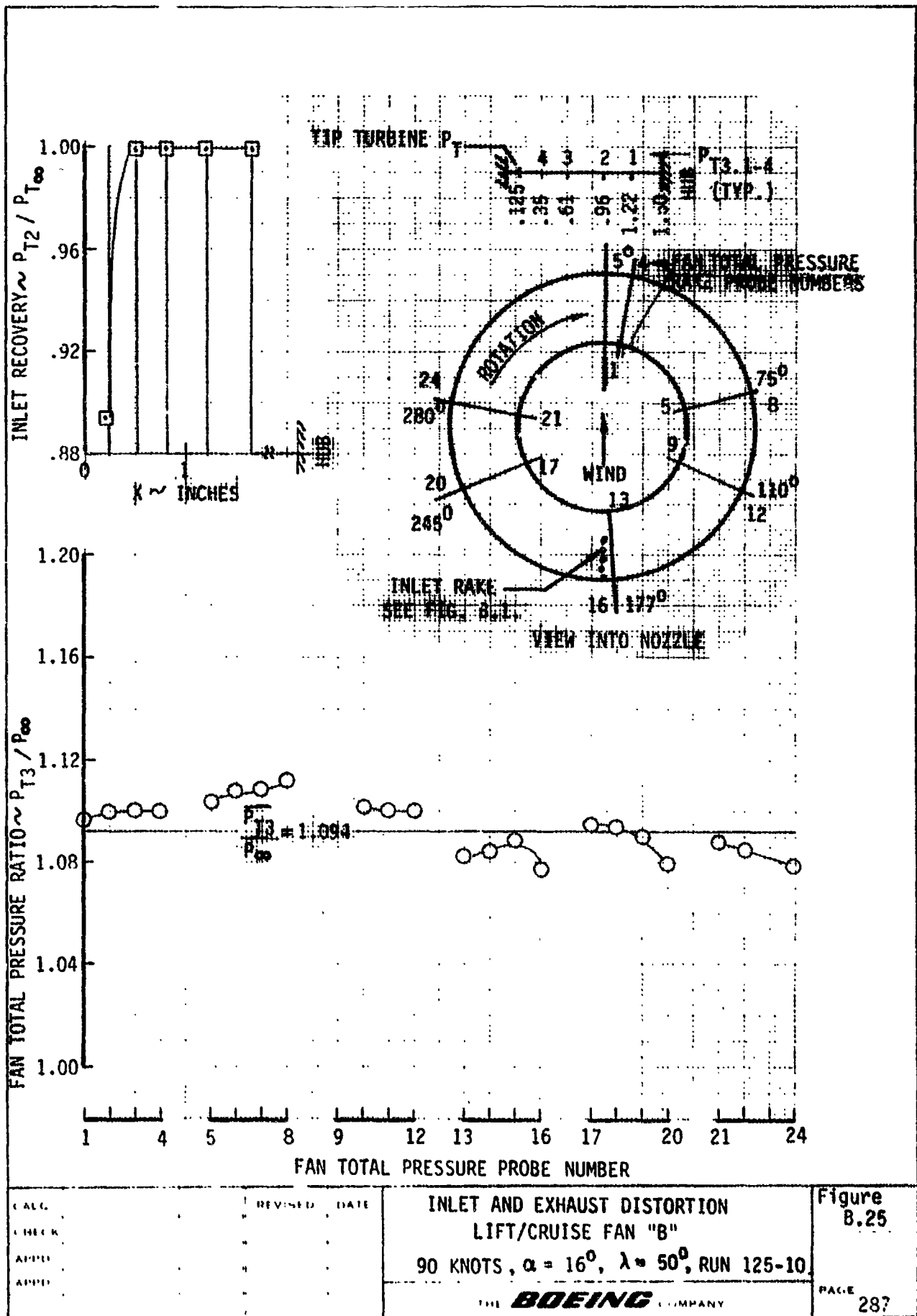
INLET AND EXHAUST DISTORTION
LIST/CRUISE FAN "B"

120 KNOTS, $\alpha = 8^\circ$, $\lambda = 90^\circ$, RUN 69-6

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Figure
B.24

PAGE 286



C4

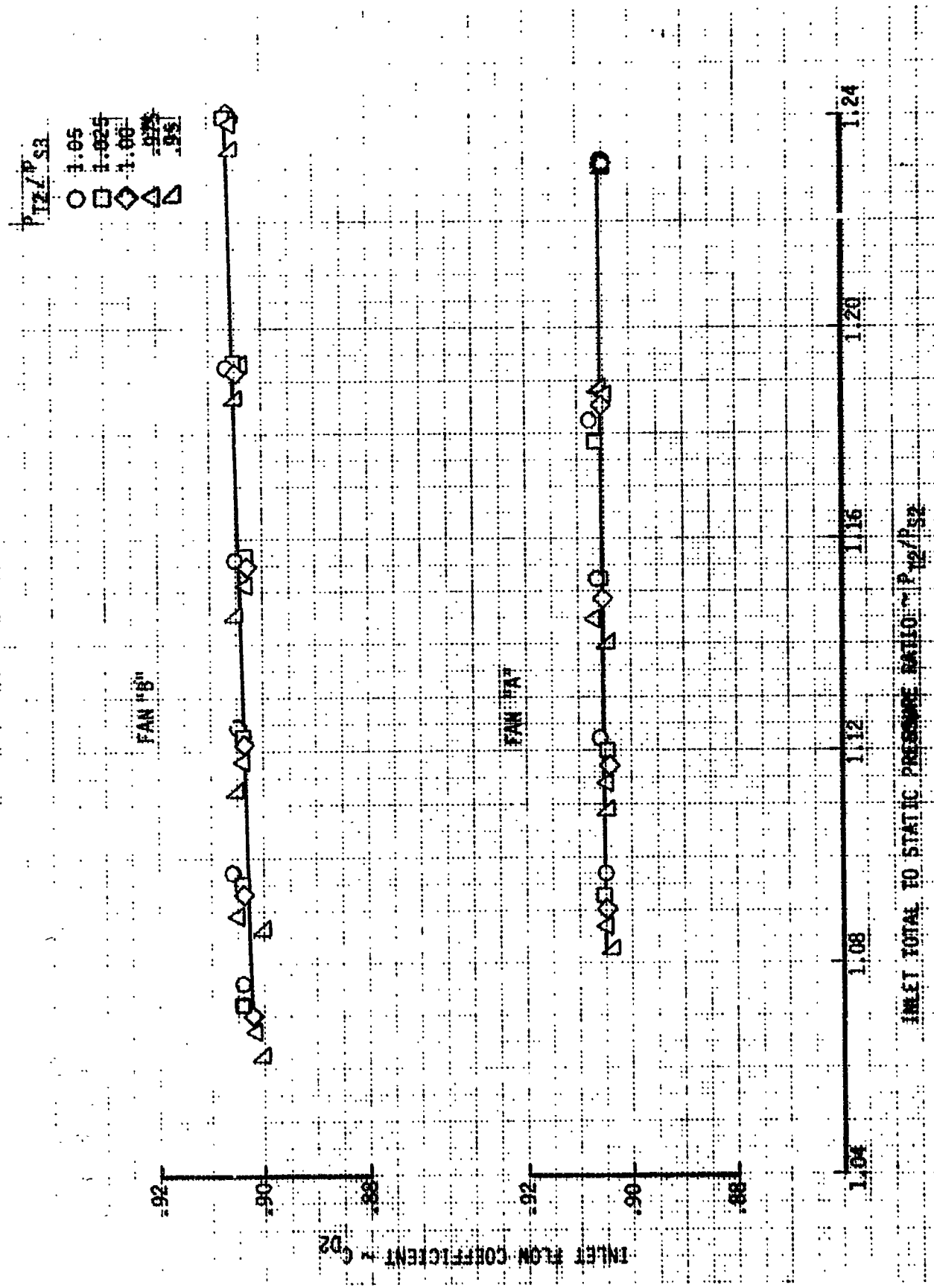
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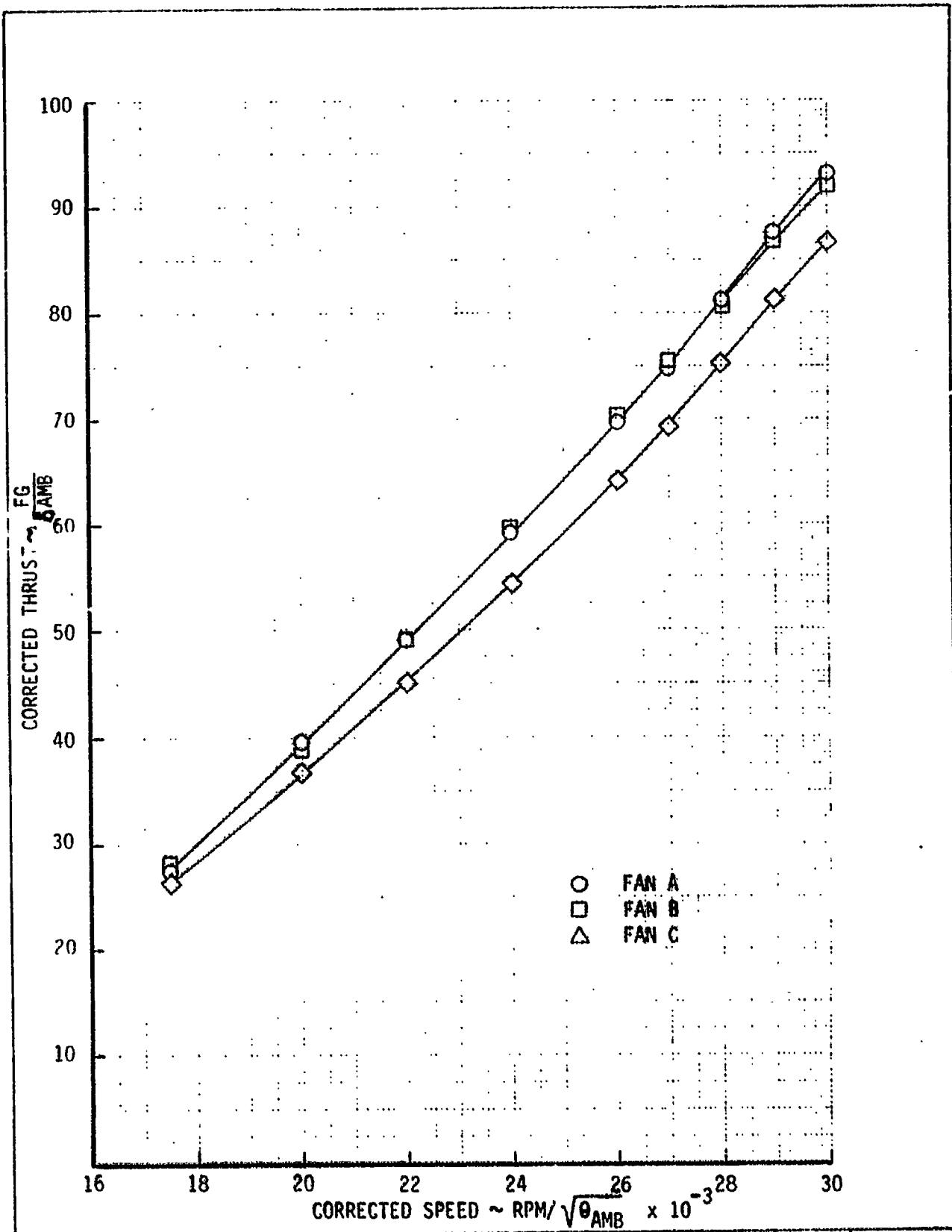
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INLET FLOW COEFFICIENTS

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Figure B.26





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APPENDIX C
DATA REDUCTION

APPENDIX C

DATA REDUCTION

In general, the force data presented in this report have had direct thrust and ram forces removed. Figure C.1 illustrates this procedure for a typical run. Direct thrust and ram forces were also removed from the three force components not illustrated. There are several reasons for removing these forces from the data as summarized below.

- It is not always possible to exactly balance the two lift/cruise fans. This can introduce rolling and yawing moments which are best removed to reveal the aerodynamic contributions.
- The nose fan thrust was not exactly constant with angle of attack. This introduces a variation of pitching moment with angle of attack which is best removed to reveal the aerodynamic forces.
- The ram force at a given C_j is not exactly representative of the airplane value. This occurs basically because of the mass flow added to drive the tip turbine and because of exhaust temperature differences between the model primary and the actual engine exhaust. Removing ram forces from the wind tunnel data facilitates correction to actual airplane ram conditions.

Removal of ram moments requires knowledge of the effective point of application of the ram forces on the model. For this purpose the results of a test of an isolated nacelle with 1.4 meter (55 inch) diameter Q-Fan installed were utilized. This test, reported in Reference 10, was run in the Ames 40'x 80' Wind Tunnel. As shown in Figure C.2, the results indicate an effective point of application which is well ahead of the inlet face plane. A constant value, $L/D_{HI} = .75$, was assumed for reducing the present data. The values of ram force for the model are summarized in Figure C.3. It is seen

that for a given fan pressure ratio (i.e., a given thrust setting) the value $CFR/\sqrt{C_J}$ is relatively insensitive to C_J (forward speed variation) and α . However, for the runs which were made at a lower thrust setting the value is somewhat different.

The gross thrust and ram force of each fan were calculated using equations C.1 and C.2. The definitions, coefficient values, and instrumentation locations associated with these equations are found in the description of the fan calibration, Appendix B.

$$F_G = C_{VF} (M_{TT} V_{ITT} + M_F V_{IF}) + C_{VPR} M_{PR} V_{IPR} \quad C.1$$

$$F_R = V_{FS} C_{DF} M_{IF} \quad C.2$$

For the purpose of removing direct thrust from the measured forces the following equations define the thrust contribution to each balance component. Note that these forces were removed from balance axis system data. The subscripts A, B, and C identify the individual simulators. Note that Fan A is the right L/C fan, Fan B is the left L/C fan, Fan C is the nose fan.

$$\text{Normal Force} = F_C \sin 76^\circ + F_B \sin \lambda_B + F_A \sin \lambda_A \quad C.3$$

$$\text{Axial Force} = -F_C \cos 76^\circ - F_B \cos \lambda_B - F_A \cos \lambda_A \quad C.4$$

$$\begin{aligned} \text{Pitching Moment} = & -\Delta x_C F_C \sin 76^\circ - \Delta z_C F_C \cos 76^\circ \\ & -\Delta x_B F_B \sin \lambda_B - \Delta z_B F_B \cos \lambda_B \\ & -\Delta x_A F_A \sin \lambda_A - \Delta z_A F_A \cos \lambda_A \end{aligned} \quad C.5$$

$$\text{Rolling Moment} = -\Delta y_B F_B \sin \lambda_B - \Delta y_A F_A \sin \lambda_A \quad C.6$$

$$\text{Yawing Moment} = -\Delta y_B F_B \cos \lambda_B - \Delta y_A F_A \cos \lambda_A \quad C.7$$

$$\text{Side Force} = 0 \quad C.8$$

The geometric constants used in the above equation were used to locate a point on the fan thrust axis relative to the moment reference center (MRC). (The reference system is shown in Figure C.5.)

$$\Delta x = \text{Fuselage Station} - (\text{Fuselage Station})_{\text{MRC}}$$

$$\Delta y = \text{Butt Line} - (\text{Butt Line})_{\text{MRC}}$$

$$\Delta z = \text{Water Line} - (\text{Water Line})_{\text{MRC}}$$

FAN	$\Delta x \sim \text{INCH}$	$\Delta y \sim \text{INCH}$	$\Delta z \sim \text{INCH}$
A	8.656	7.482	3.09
B	8.656	-7.482	3.09
C	-20.215	0	-2.369

The following equations were used to calculate the components of ram force to be removed from the measured force data. These increments were subtracted from the stability axis data before application of the wind tunnel wall effects corrections. The axis system is shown in Fig. C.5.

$$D_R = \text{Ram Drag} = F_R \cos \beta \quad \text{C.9}$$

$$SF_R = \text{Ram Side Force} = -F_R \sin \beta \quad \text{C.10}$$

$$\text{Ram Pitching Moment} = U_R (\Delta z \cos \alpha - \Delta x \sin \alpha) \quad \text{C.11}$$

$$\text{Ram Rolling Moment} = SF_R (\Delta z \cos \alpha - \Delta x \sin \alpha) \quad \text{C.12}$$

$$\text{Ram Yawing Moment} = -SF_R (\Delta x \cos \alpha + \Delta z \sin \alpha) + D_R \Delta y \quad \text{C.13}$$

The constants (Δx , Δy , and Δz) for this set of equations are as follows:

FAN	$\Delta x \sim \text{INCH}$	$\Delta y \sim \text{INCH}$	$\Delta z \sim \text{INCH}$
A	$8.656 - 10.673 \cos \lambda$	7.482	$3.09 + 10.673 \sin \lambda$
B	$8.656 - 10.673 \cos \lambda$	-7.482	$3.09 + 10.673 \sin \lambda$
C	-22.02	0	5.078

The ratio of free stream to jet efflux velocity (V/V_J) is defined as:

$$V/V_J = \sqrt{\frac{2 A_T}{C_J S}} = \frac{.561}{\sqrt{C_J}} \quad \text{C.14}$$

where

S is the model reference area

C_J is the thrust coefficient F_G/qS

F_G is the gross thrust as defined in equation C.1

q is free stream dynamic pressure

A_T is the propulsive exit area with appropriate adjustments such that V_J is representative of fan efflux velocity rather than a weighted average of the fan efflux and blown primary velocities.

Wall Corrections

Corrections for tunnel wall constraints were determined based on Heyson's Interference Theory (Reference 11. and 12.). In this theoretical approach, a lifting surface is represented by a number of lifting line elements oriented along the quarter chord line and trailing along the wake. Ten lifting elements were chosen for the T-39 model. The effective wake deflection angle was determined as discussed in Reference 13. Corrections were determined separately for the wing and the tail. The total model loads (lift and drag) as measured directly by the internal balance (i.e., including direct thrust and ram forces) were used in determining the interference velocities.

The corrections were computed in the form of a flow angle change, which changes the resolved lift and drag components, and a tunnel test section dynamic pressure change. In addition to the above corrections, a conventional solid blockage correction factor was applied to the dynamic pressure.

Figure C.4 presents a summary of typical wall correction factors computed at the wing where $\Delta\alpha_w$ denotes the computed change in stream direction at the wing and Q/Q_U denotes the ratio of the final corrected stream dynamic pressure to the value before application of wall corrections.

The pitch center was not located at the model center with the result that the model changed height in the tunnel as angle of attack was varied. At $\alpha = -8^\circ$ the model moment reference center (MRC) was 8.5 inches below tunnel center line while at $\alpha = +30^\circ$ the MRC was 15.5 inches above tunnel center line. The height variation with alpha was linear. This variation was taken into account in the wall effects calculations.

Figures A.18 thru A.21 in Appendix A, present some data with the ground plane installed. These data were corrected for the wall effects of three walls only with the ground effect left in.

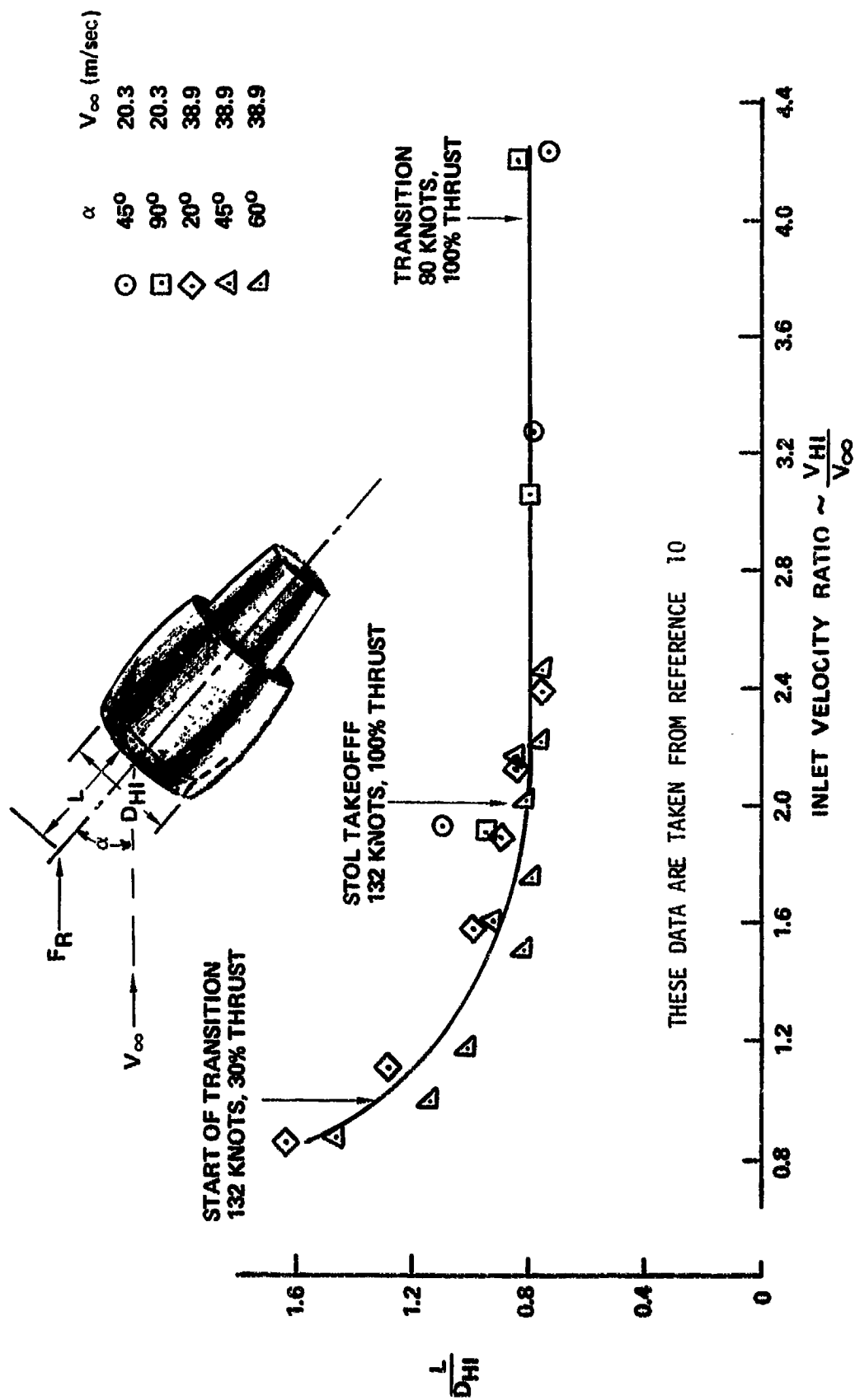


FIGURE C.2 LOCATION OF RAM DRAG RELATIVE TO INLET PLANE

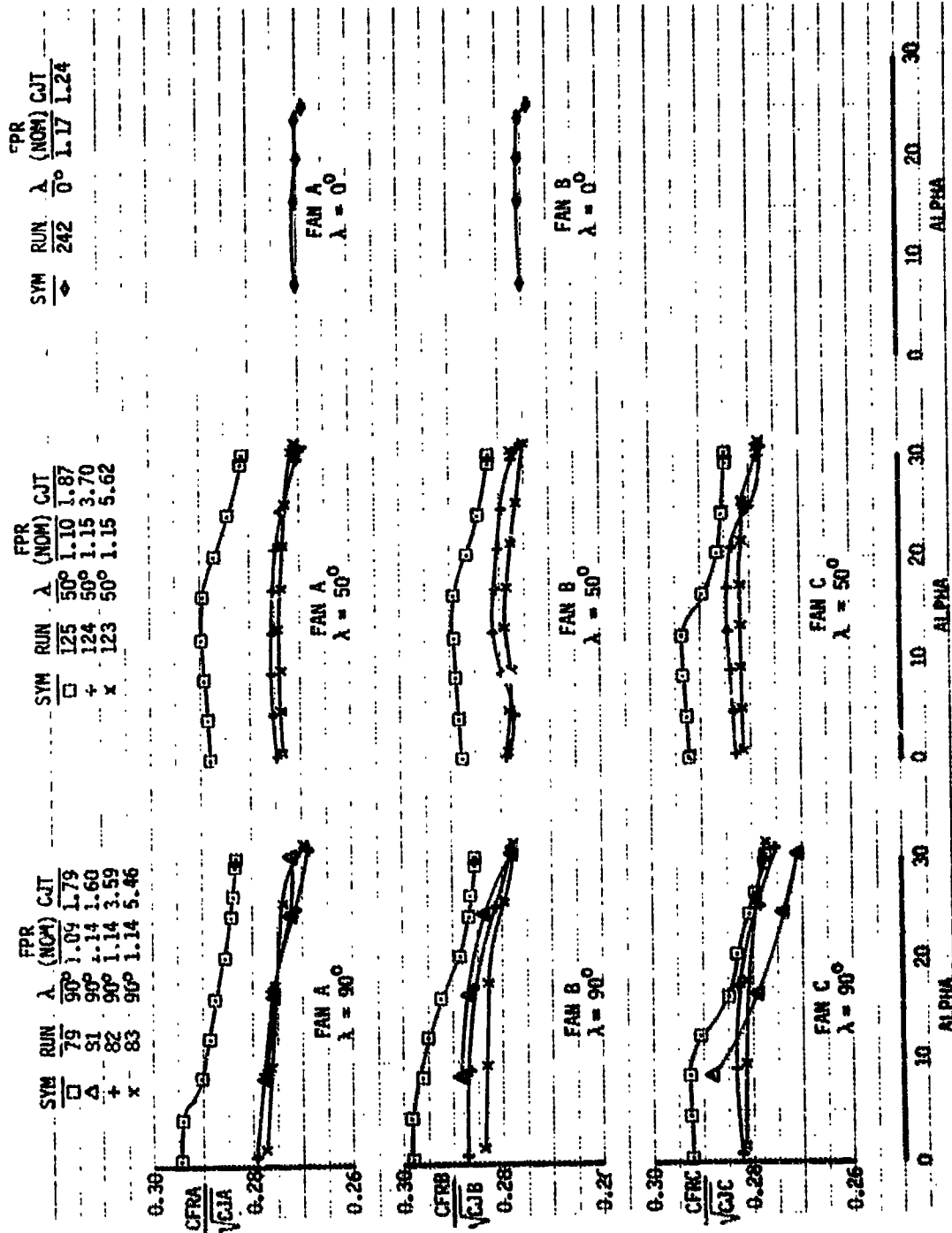
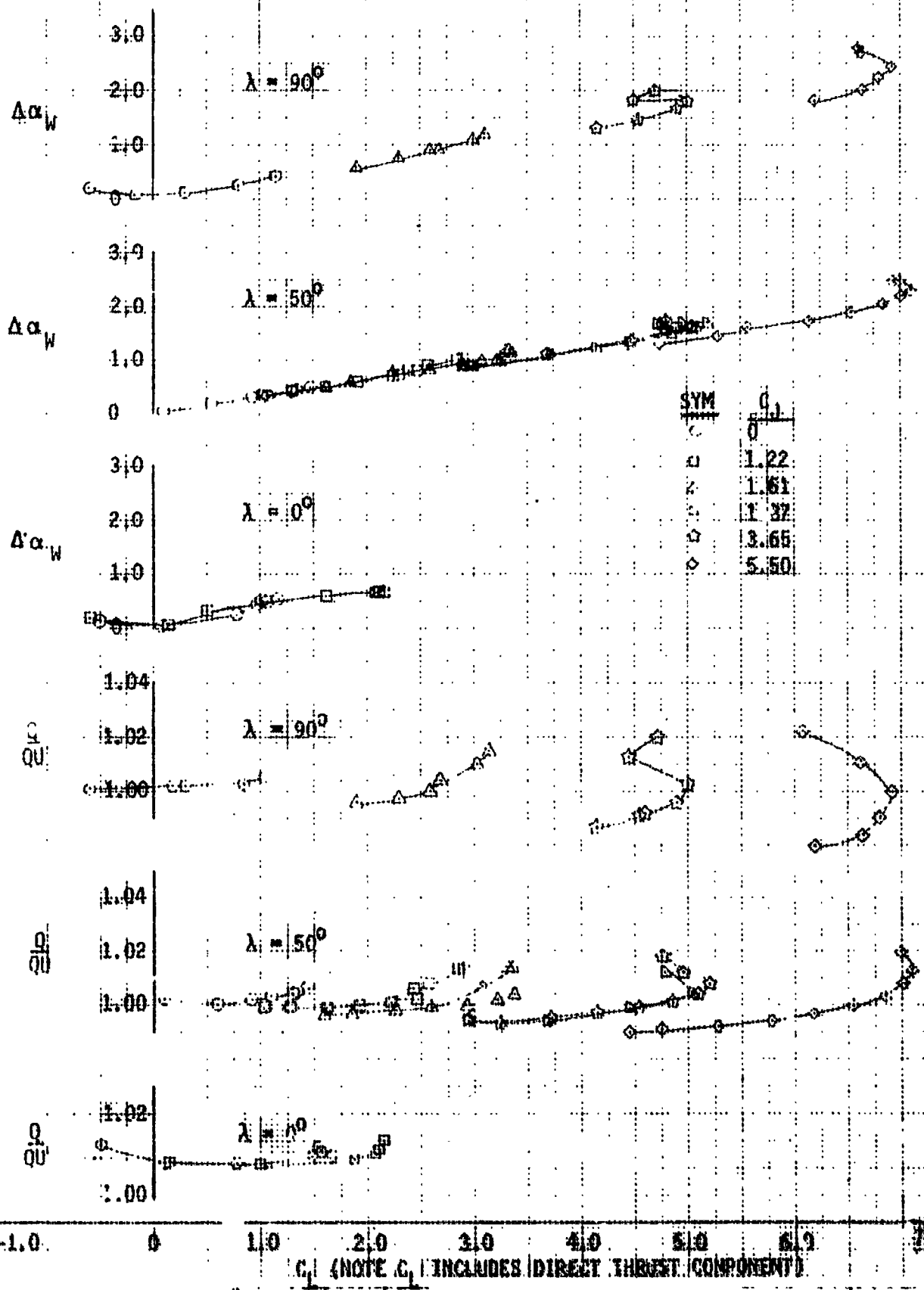
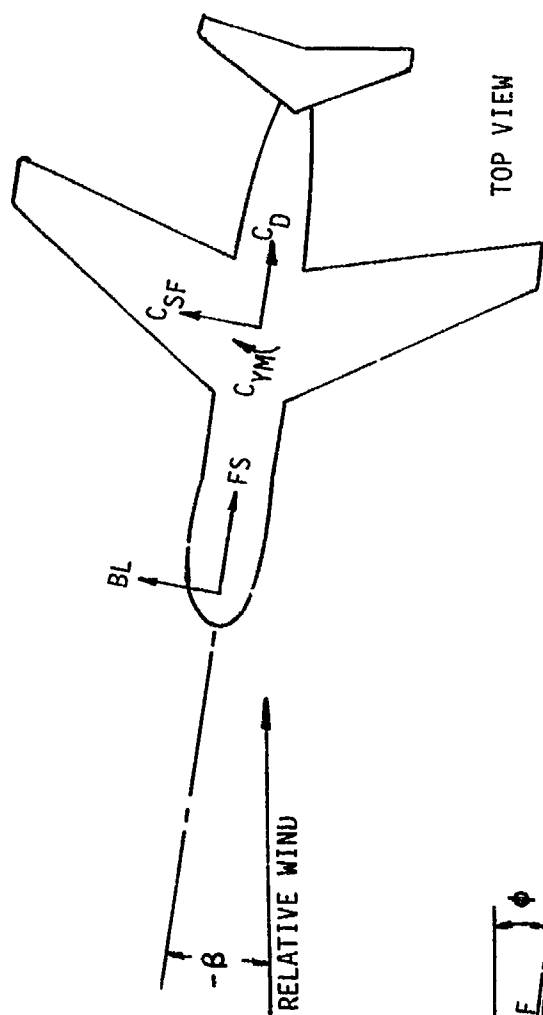
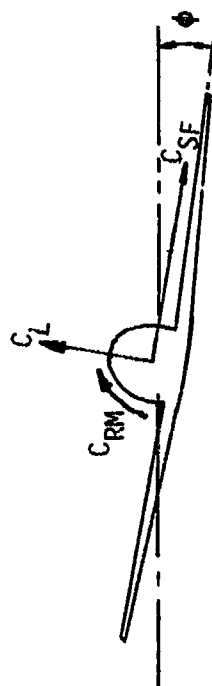


FIGURE C.3 SUMMARY OF TYPICAL RAM DRAG DATA

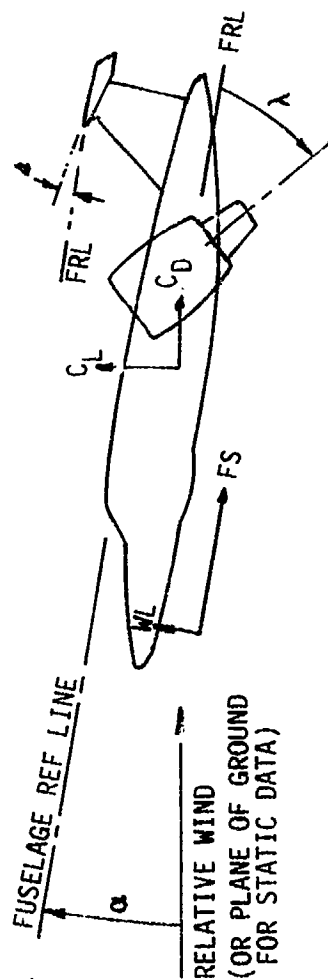




TOP VIEW



REAR VIEW



FS = FUSELAGE STATION

BL = BUTT LINE

WL = WATER LINE

FIGURE C.5 SIGN CONVENTION